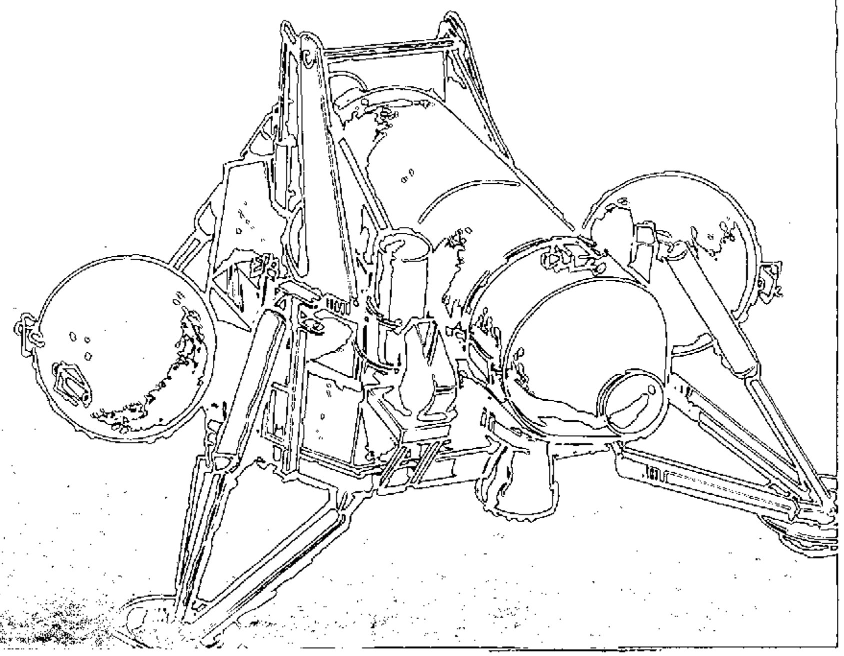
March 1974

Midterm Review

A Feasibility Study of Unmanned Rendezvous and Docking in Mars Orbit



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A FEASIBILITY STUDY OF UNMANNED RENDEZVOUS AND DOCKING IN MARS ORBIT

Mid-term Review

March 1974

Prepared Under JPL Contract 953746

JPL Technical Manager - Jesse W. Moore

Approved

W. T. Scofield
Program Manager
Advanced Planetary Programs

FOREWORD

This document contains copies of the visual aids used in the mid-term presentation of "A Feasibility Study of Unmanned Rendezvous and Docking in Mars Orbit" (JPL Contract 953746). It is submitted in response to Article 1, Paragraph (a), (2), (B) of the Contract Schedule. The oral presentation was made by Martin Marietta Corporation at the Jet Propulsion Laboratories on March 1, 1974.

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SUMMARY

AND

MISSION PERFORMANCE

Performance Analysis - S. K. Asnin

W. T. Scofield

SCIENCE OBJECTIVES FOR MARS SAMPLE RETURN

Many of the questions that scientists have about the origin, evolution and present state of Mars can be answered only by highly sophisticated and carefully controlled investigations. Such investigations, examples of which are listed here, can best be done in Earth laboratories.

Age Dating determines when material in the lithosphere was solidified, when and how often it has been remelted and how long it has been on the surface (e.g., cosmic ray exposure history).

Impact History records the relative chronology of surface formations and calibrates episodes of meteor bombardment. Such episodes can be correlated with Earth and Moon data to develop a larger perspective on the history of our solar system.

Geochemical Constitution reveals valuable insights into the evolution of the planet through the measurement of the types and abundances of trace elements and the submicroscopic distribution materials in general.

Mineral Assemblages and Relationships tell the story of the planet's acretion processes and the metamorphoses that have occurred since.

Radioactive Element Content measures the differentiation processes that have been active in the planet's history and contributes powerful inferences about the constitution of the mantle and core.

Oxidation States and Trapped Gases record the history of the interaction of the surface and the atmosphere.

Remanent Magnetization tells about past magnetic fields (at the time of crystalization) and records clues to plate tectonic activity (continental drift).

Organic Analysis can differentiate between biologically and non-biologically derived organic compounds and can make paleontographical surveys (search for fossils as evidence of past life forms).

Life Detection and Analysis are potentially the most dramatic and exciting of the scientific investigations that can be performed on returned Mars samples. Life forms exhibiting basic processes different from our own can perhaps only be detected and understood through extremely careful Earth laboratory work.

SCIENCE OBJECTIVES FOR MARS SAMPLE RETURN

Age Dating

Impact History

Geochemical Constitution

Mineral Assemblages and Relationships

Radioactive Element Content

Oxidation States and Trapped Gases

Remanent Magnetization

Organic Analysis

Life Detection and Analysis

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SCIENCE REQUIREMENTS ON MARS SAMPLE ACQUISITION

The requirements imposed by the science investigators involved in an MSSR mission that will directly affect the mission and spacecraft design are summarized here. Not all of these requirements would be met in a minimum MSSR mission.

In order to completely satisfy the diversity of samples and sampling locations, multiple sampling devices and perhaps a rover would be required.

The experience of the Russian Luna 16 mission has established the adequacy of small samples for doing even very sophisticated analysis.

Sample documentation is required for making decisions on the samples to be taken and for input data to the sample analysis (e.g. knowledge of the orientation of the sample on the surface is vital to the interpretation of remanent magnetization measurements).

Sample protection must not only preserve environmental conditions and keep out alien material but must also prevent possible reactions between the sample and the sample canister material.

SCIENCE REQUIREMENTS ON MARS SAMPLE ACQUISITION

Samples from Several Locations to Include:
- Surface Dust
- Soil Core Tube
- Bedrock Drill Chips
- Loose Rocks
- Atmosphere

At Least One Gram from Each Location

Sample Documentation:
- Teleimagery
- Elemental Analysis
- Film Photo
- Meteorological Conditions

Sample Protection:
- Vacuum Seal
- Temperature Control
- Contamination Control

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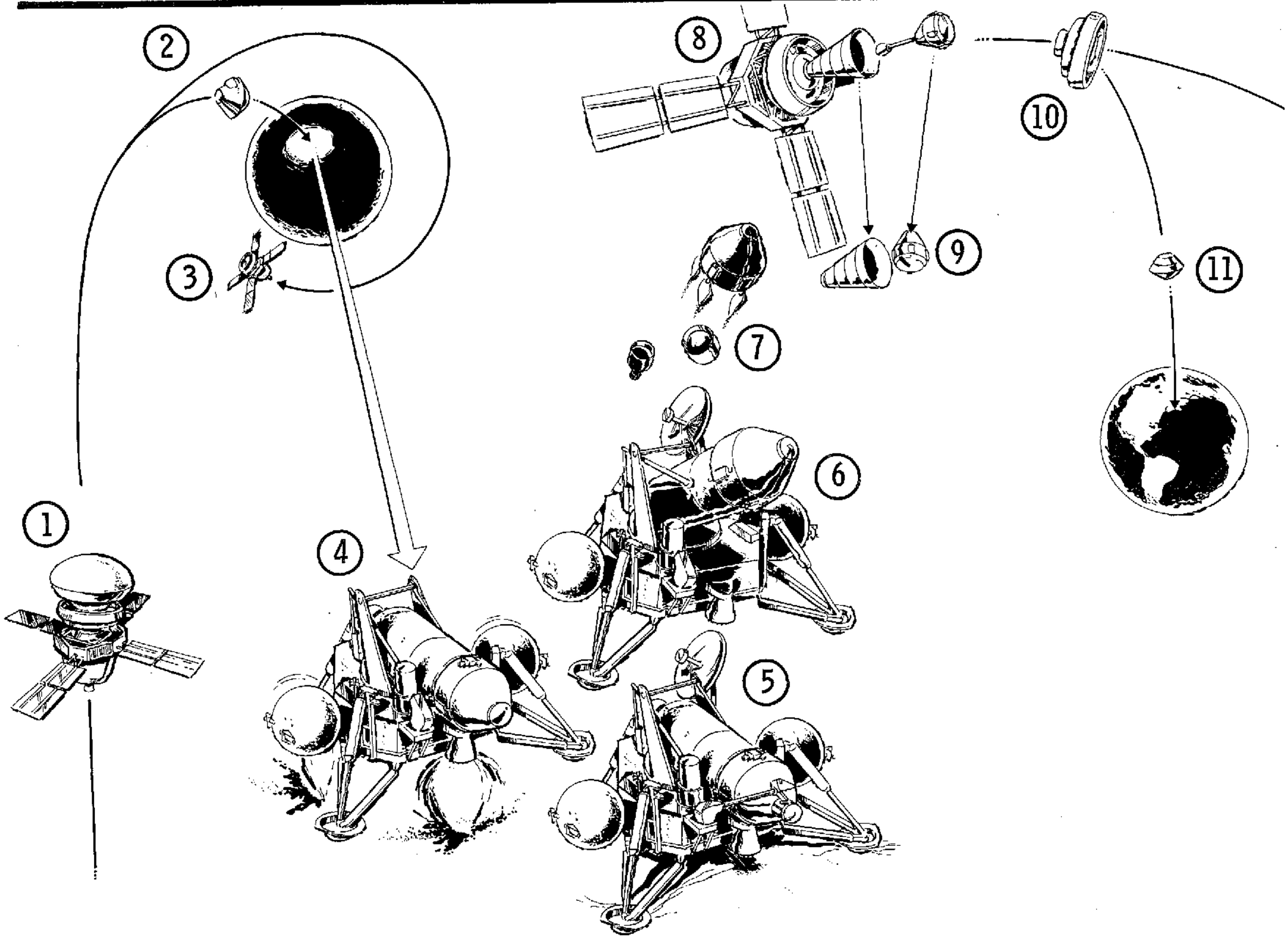
MSSR MISSION SEQUENCE - MARS RENDEZVOUS MODE

This is a typical mission sequence for a Mars sample return using the Mars orbital rendezvous mode. The numbers on the drawing refer to the following events:

1. Earth launch and cruise to Mars of the total spacecraft comprising the orbiter, lander, Mars ascent vehicle (MAV) and Earth return vehicle (ERV).
2. Lander (with MAV) separates and performs a direct entry from the incoming asymptote.
3. Orbiter (with ERV) goes into Mars orbit.
4. Lander lands.
5. Sample collected and stowed on MAV sample canister.
6. MAV erected and launched.
7. MAV stages and injects into rendezvous orbit.
8. Rendezvous, docking and sample transfer.
9. Docking cone and MAV discarded.
10. ERV injected to Earth return trajectory.
11. Earth entry capsule separated for entry and recovery.

MSSR MISSION SEQUENCE - MARS RENDEZVOUS MODE



PRINCIPAL MSSR MISSION CONCERNS

Studies and deliberations on the Mars sample return conducted by NASA, industry and the scientific community have all concluded that these three issues must be faced and dealt with before a decision to proceed with the mission can be made. Back contamination concerns, or the potential danger that returned Mars biota could have pathogenic or unbalancing effects on the Earth's biosphere, are being studied at the present time under the direction of NASA Headquarters' exobiology office.

The study being reported on here is examining what appears to be the major technical feasibility concern in the mission, that of the ascent rendezvous, docking and transfer of the sample at Mars.

The potential runout cost of the mission can only be calibrated after the first two issues are better understood. At the present time, cost estimates have varied from the order of half a billion to several billion dollars.

PRINCIPAL MSSR MISSION CONCERNS

Back Contamination

Technical Feasibility

Cost

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OBJECTIVES OF THE URDMO STUDY

This study has the primary objective of investigating the ascent, rendezvous, docking and sample transfer operations in a potential MSSR mission that uses the Mars orbital rendezvous mode. In order that the design choices made for these operations remain compatible with the rest of the mission, the impact on the Earth launch, Mars landing and orbiting and Earth return phase are also being assessed in a cursory manner.

The approach to the study has involved the selection and description of a preliminary baseline concept that will be presented at the mid-term review. Mr. J. W. Moore, JPL Technical Manager, has participated in and approved the preliminary baseline choices. The second half of the study will be an examination of alternatives to the baseline features or more in depth analysis of those features that appear to warrant it.

OBJECTIVES OF THE URDMO STUDY

1. Assess the Technical Feasibility of:

   Mars Ascent

   Mars Orbital Rendezvous

   Automatic Docking and Sample Transfer

2. Test the Fit of the Above Functions with:

   Earth Launch

   Mars Landing and Orbiting

   Earth Return

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CURRENT MISSION BASELINE (MARCH 1974)

The baseline mission being described in this mid-term presentation includes the features listed here. Some of the more important decisions made in selecting this baseline involved the following reasoning:

1. 1981 is the earliest conceivable mission year. The next available opportunity (1983/84) poses more difficult performance problems, but, as it works out, the baseline described here could be performed in 1983/84 if the orbiter propulsion system were converted to space storable propellants.

2. The nominal 20-day launch period was arrived at after consultation with NASA's Lewis Research Center.

3. The direct entry lander concept is based on rather extensive work done in 1970 under the Viking project in a study known as the Option B Concept.

4. The 4° entry corridor is a compromise choice that eliminates the need for optical approach guidance and allows alignment of the incoming and outgoing asymptotes in the same plane. More landed weight performance could be achieved by going to a 2° entry corridor.

5. The 2200 km altitude for the rendezvous orbit results from a tradeoff among the performance requirements of all the spacecraft elements (launch vehicle, lander, orbiter, MAV, and Earth return vehicle).

CURRENT MISSION BASELINE (MARCH 1974)

1. 1981 Mission
2. Single Titan IIIE/Centaur Launch
3. 20-Day Launch Period
4. Direct Entry Lander (Modified Viking '75)
5. 4^0 Entry Corridor
6. Rendezvous Orbit Plane Contains Incoming/Outgoing Asymptotes
7. 2200 km Rendezvous Orbit (Circular)
8. Three Stage MAV (Solid, Solid, Liquid)
9. Three Axis Stable MAV
10. Separate ERV (Pioneer Venus Derivative)
11. 1 kg Sample Weight

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TYPICAL MISSION TIMELINE - 1981 MISSION

This simplified mission event sequence indicates the timing of a typical 1981 launched Mars sample return. The total time of approximately 1050 days from Earth launch to sample return is typical of the conjunction class mission.

A more detailed timeline is provided in the navigation analysis section of this presentation.

TYPICAL MISSION TIMELINE - 1981 MISSION

1. Earth Launch - November 13 - December 2, 1981
2. Mars Encounter (Lander Separation) - September 15-25, 1982
3. Mars Landing and Orbit Insertion (1000 x 100,000 km Orbit) - Mars Encounter + 4 Hours
4. MAV Launch - Mars Landing + 11 Days
5. Rendezvous, Docking and Sample Transfer - MAV Launch + 16 Days
6. ERV Inject to Earth Return Trajectory - Sample Transfer +~400 Days (November 19-28, 1983)
7. Earth Arrival - September 28 - October 1, 1984.

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EARTH LAUNCHED PAYLOAD

The configuration of the current baseline MSSR spacecraft is outlined in this drawing. The concept emphasizes the use of existing technology, specifically Viking and Pioneer Venus.

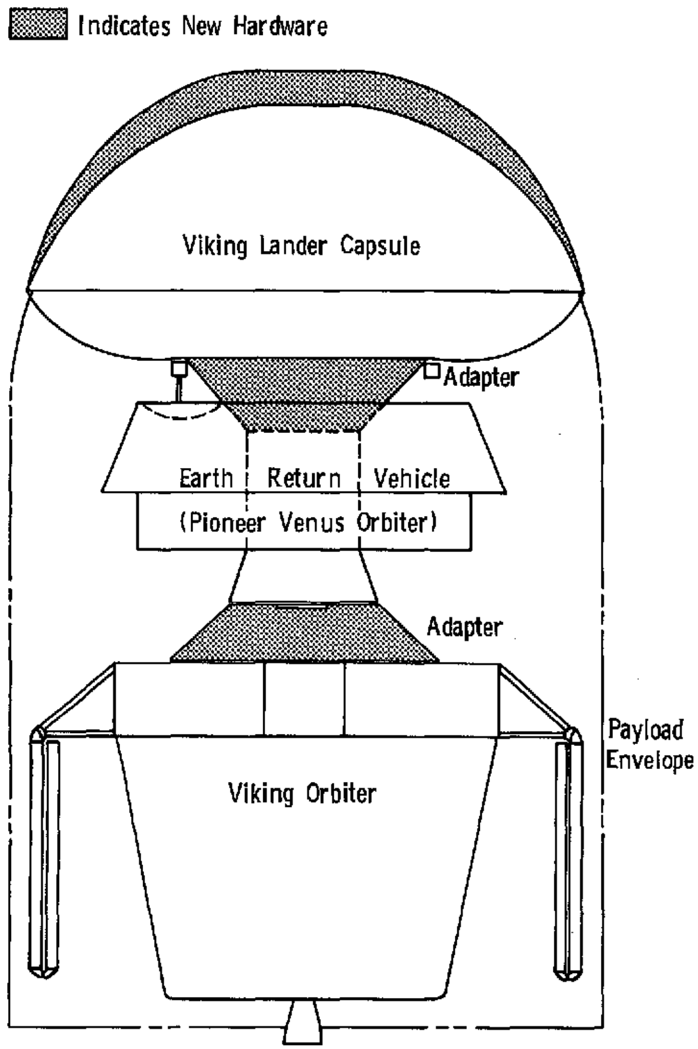
The Viking Orbiter propellant capacity is increased by 20% over the nominal VO'75 loading (1405 to 1692 kg).

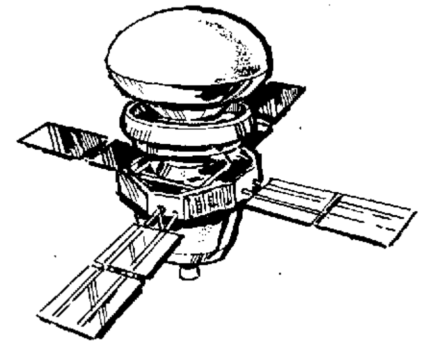
The Earth Return Vehicle (ERV), adapted from the Pioneer Venus spacecraft in this case, is mounted between the lander and orbiter.

The Viking Lander Capsule is enlarged by the amount shown to accommodate the Mars Ascent Vehicle (MAV).

Total spacecraft injected weight is 4244 kg which includes a project reserve of 41 kg. This compares with the Viking '75 spacecraft injected weight of 3500 kg.

EARTH-LAUNCHED PAYLOAD





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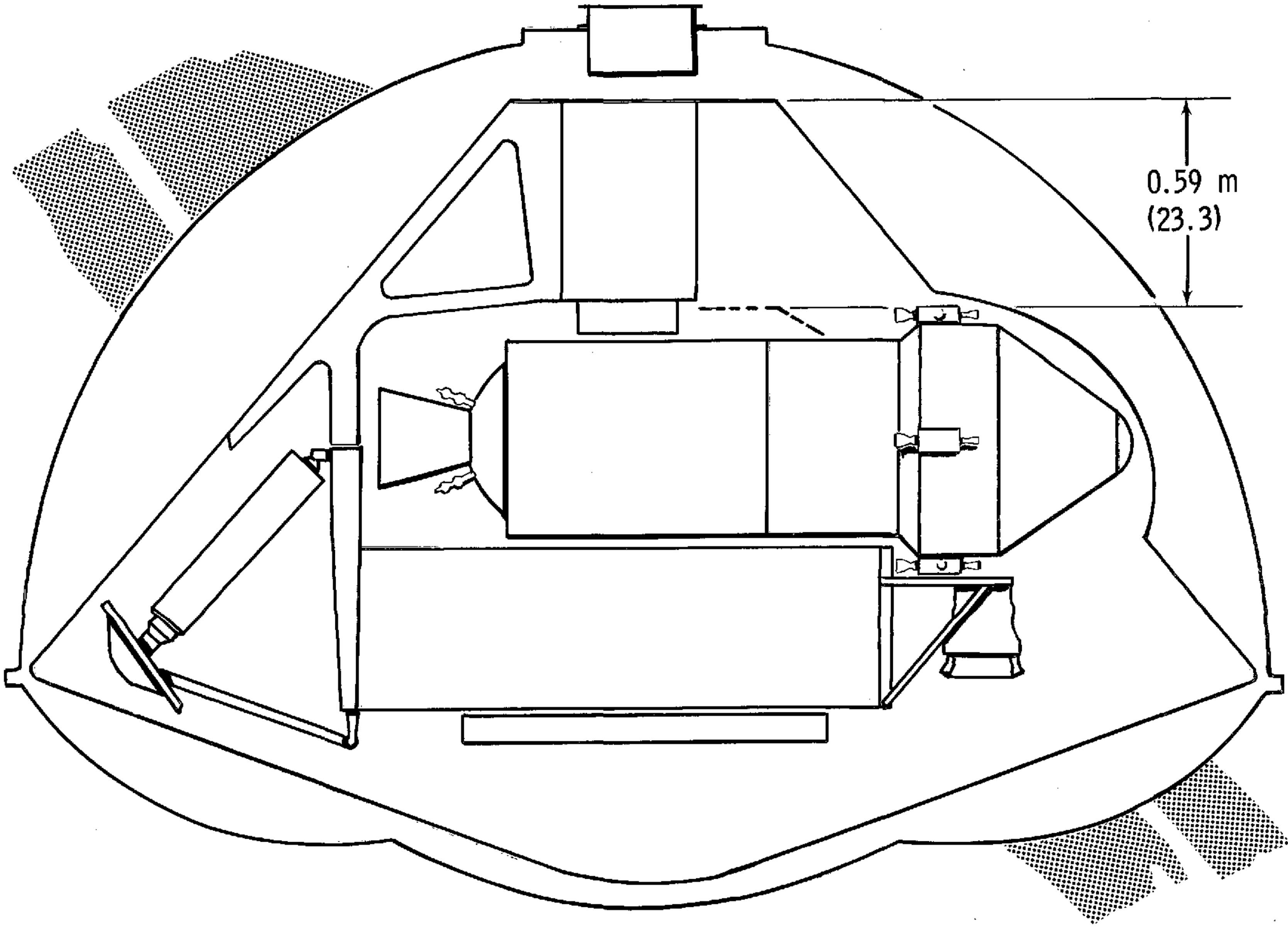
MAV IMPACT ON VIKING LANDER CAPSULE

This illustration shows the accommodation required in the lander capsule for the MAV.

The parachute canister is raised 59 cm and a new parachute support truss provided. The aeroshell aft body and the bioshield base will also be redesigned.

The direct entry mode will necessitate a beef-up of the heat shield and support structure, compared with Viking '75. Entry velocity increases from approximately 4628 mps (15,184 fps) to 5785 mps (18981 fps).

MAV IMPACT ON VIKING LANDER CAPSULE



LANDER MODIFICATIONS

The changes to the Viking Lander landed configuration required to mount the MAV and its launcher are shown here.

All lander science, except one camera, is removed. The two Snap-19 (35 watt) RTGs are replaced by two later model Teledyne 20 watt units. The lander telecommunications systems (S-Band and UHF) are removed and replaced by a modified MAV S-Band system.

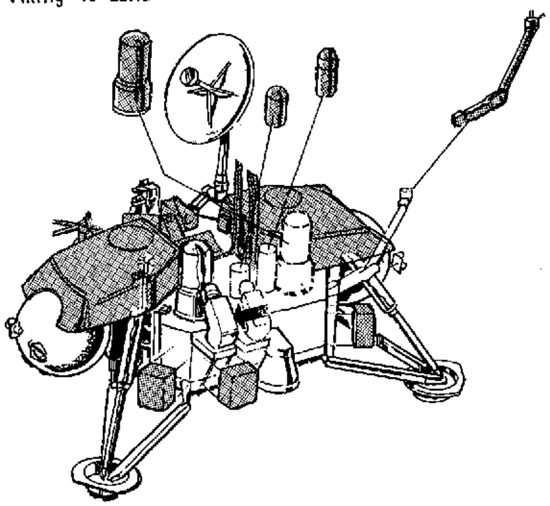
The MAV launcher is mounted on the lander equipment plate (with appropriate load carrying stiffeners added) and provides 360^{o} of azimuth rotation and 79^{o} of elevation.

The lander terminal descent propulsion system is modified to carry 75 kg of propellant. This requires the addition of an external pressurization sphere and regulator.

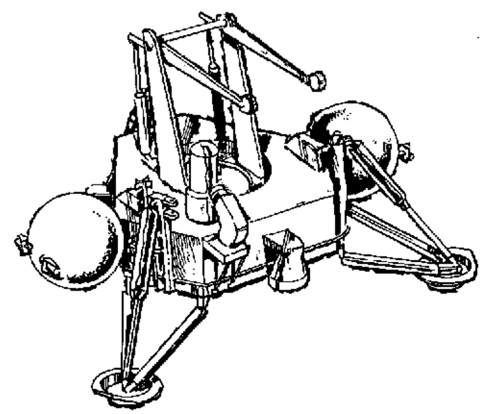
Total landed weight of this configuration is 773.6 kg (1705.5 lbs) compared with the Viking '75 landed weight of 594.2 kg.

LANDER MODIFICATIONS

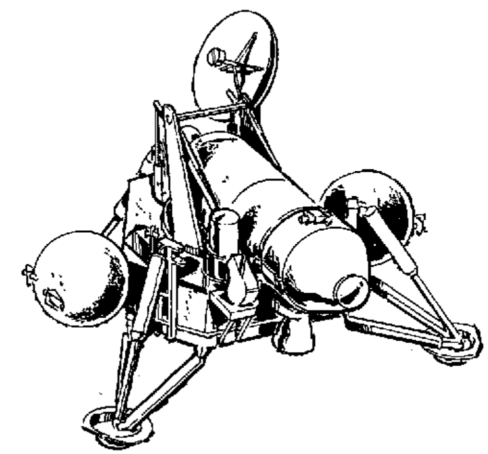
Viking '75 Land



Modified Lander



Lander With MAV



Indicates Components Not Required for Sample Return Mission

MARS ASCENT VEHICLE

The current baseline MAV is a three stage, three axis stable, launch vehicle weighing 290 kg (637 lbs). It is capable of automatically ascending to a 100 x 2200 km orbit and thereafter being commanded to circularize at 2200 km into the rendezvous orbit.

The MAV is the only entirely new vehicle in the MSSR spacecraft configuration. The design approach is to keep the MAV as simple as possible and keep its maneuvers under Earth or orbiter control whenever feasible.

Salient features of the MAV subsystems include:

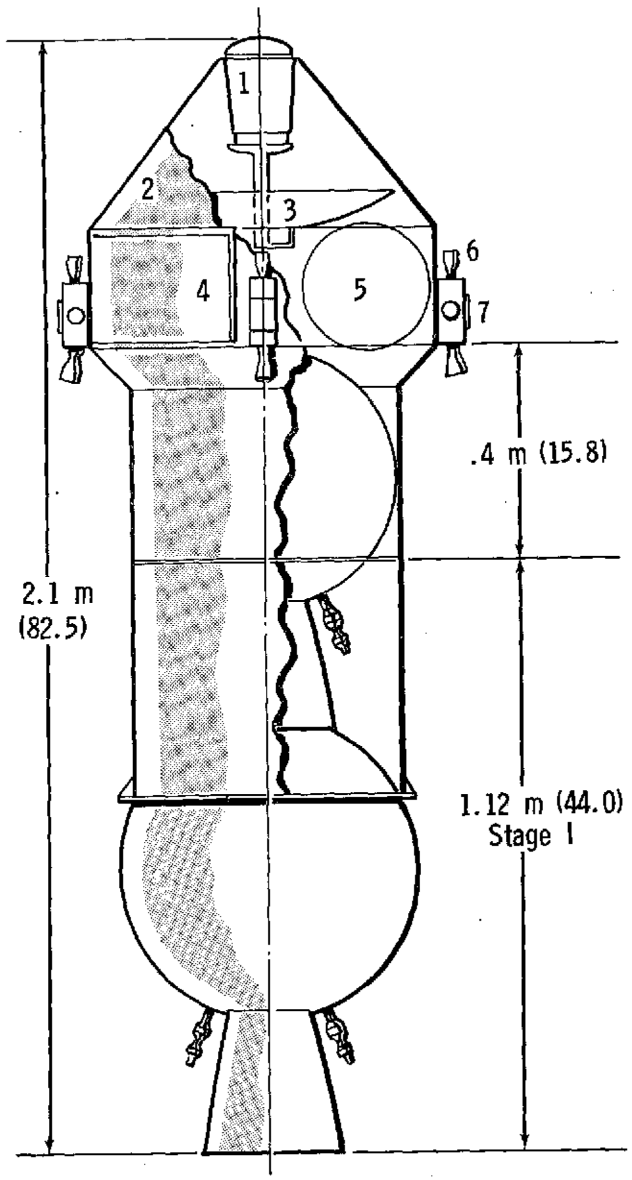
Guidance and Control - Open loop, constant pitch over rate with rate gyro reference during ascent and sun sensor/Earth pointing reference during orbital operations.

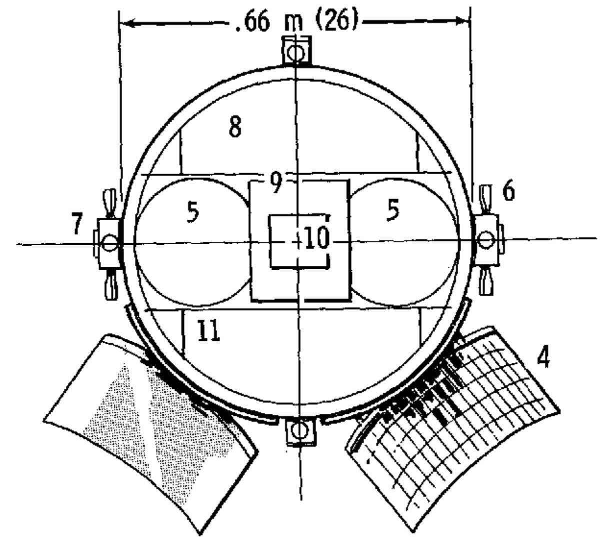
Telecommunications - S-Band, angle tracking, dual ratio transponder. Earth tracking provides command, telemetry and 2-way coherent doppler links. Orbiter tracking provides pointing reference during rendezvous. 20" high gain antenna with monopulse feed. Maximum transmitter output is 4 watts.

Propulsion - Sterilizable solid propellant Stage I and II. Monopropellant hydrazine Stage III for thrust vector control, attitude control, orbit circularization and orbit trims.

Power - Solar cells (0.11 m^2) and Ni-H_2 battery.

MARS ASCENT VEHICLE





Legend:

1. Sample Canister
2. R/F Transparent Fairing
3. Antenna
4. Solar Panel (4)
5. Stage III Propellant Tank (2)
6. ACS Motor Assembly (4)
7. Sun Sensor Assembly (4)
8. Telecommunication System
9. Antenna Electronics
10. Boom Drive Mechanism
11. Electrical & Flight Control Subsystems

CRITICAL ELEMENTS OF URDMO PROFILE

This illustration summarizes the critical questions and answers relating to whether or not the baseline configuration will perform a successful unmanned rendezvous and docking in Mars orbit.

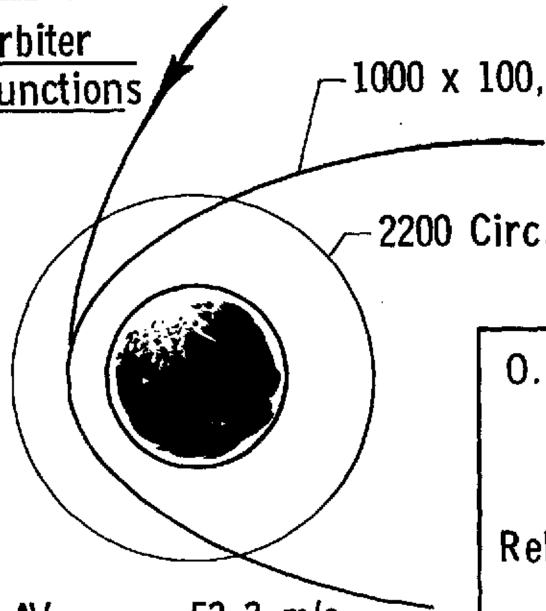
The questions addressed in our study so far are the following:

1. Can the orbiter insert into the initial capture orbit and then maneuver to the 2200 km altitude rendezvous orbit with an affordable propellant allowance for uncertainties and errors (ΔV_{stat})?
2. Can the orbital parameters of the orbiter and MAV be determined accurately enough with DSN tracking to calculate further maneuvers?
3. Can the relative state of the orbiter and MAV be determined accurately enough (using ΔVLBI tracking)?
4. Can the MAV ascend automatically and insert into a stable orbit to permit Earth-based tracking for further maneuvers?
5. Can the MAV be commanded to the 2200 km rendezvous orbit with an affordable ΔV_{stat}?
6. Can the orbiter be phased into the rendezvous orbit so that the dispersions on the separation between the orbiter and MAV can be handled within the rendezvous radar maximum range?
7. Can a rendezvous algorithm be devised that will bring the orbiter and MAV together with an affordable allocation of rendezvous propellant and affordable weight and power allocations for rendezvous hardware?

Studies to date indicate that all of these questions can be answered affirmatively.

CRITICAL ELEMENTS OF URDMO PROFILE

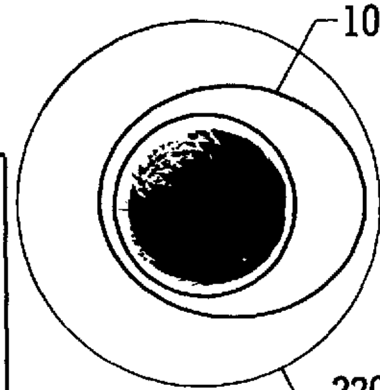
Orbiter Functions



ΔV_{STAT} = 53.3 m/s

O.D. =	2.0 km
	1.6 m/s
(DSN)	
Rel State =	0.31 km
	0.15 m/s
(ΔVLBI)	

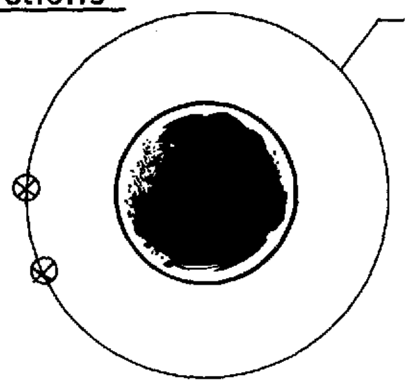
MAV Functions



MAV Ascent h_p Dispersion = 4.7 km

ΔV_{STAT} = <50 m/s

Rendezvous Functions



TRI Separation = 200 $\pm$ 50 km

ΔV_{Rend} = 61 m/s (from 250 km)

Rend. Prop. = 26.6 kg

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SAMPLE TRANSFER AND CONTAMINATION CONTROL

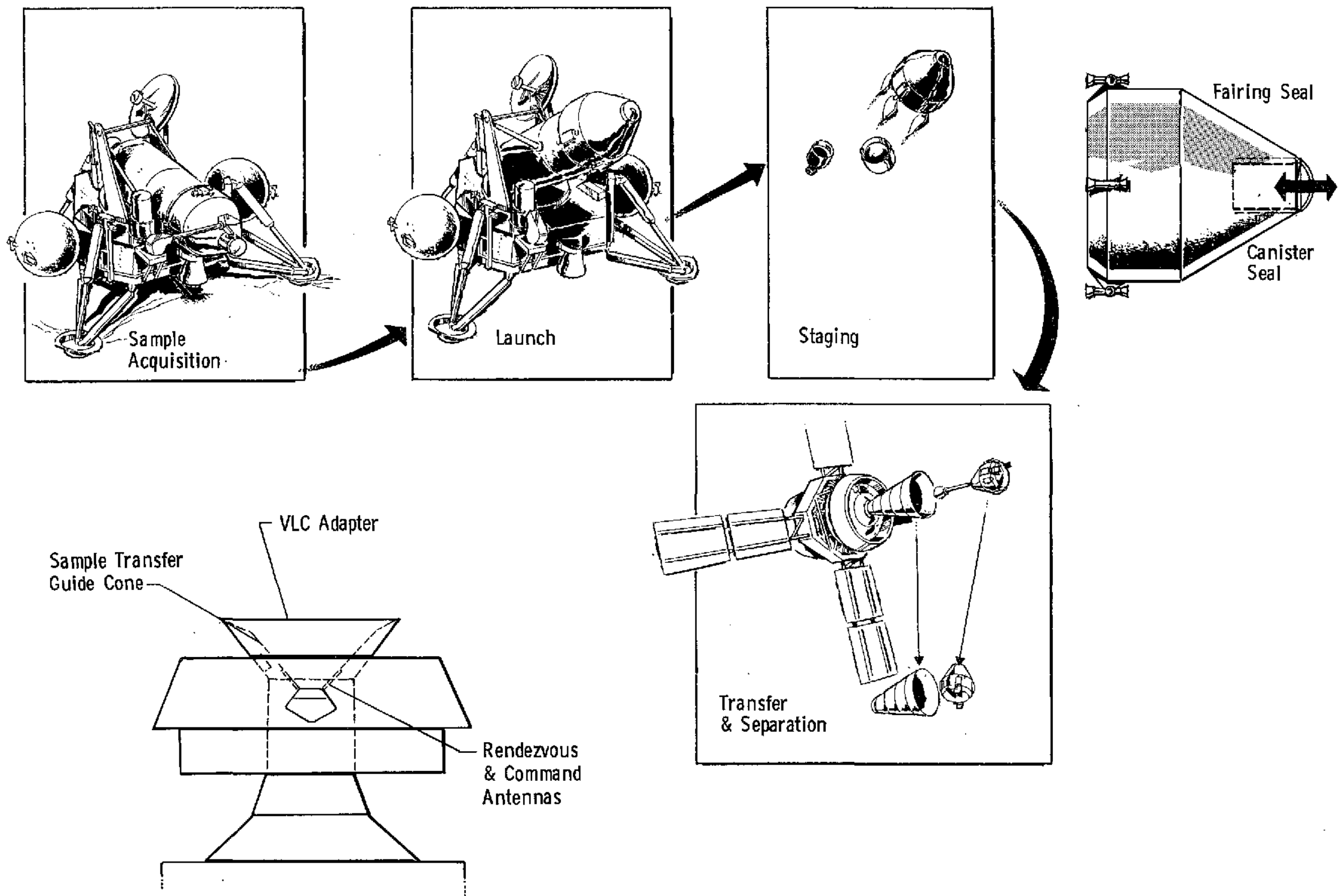
This drawing shows the sequence of sample loading, launch, rendezvous, and sample transfer and highlights the approach to minimizing the transfer of contaminents from the MAV to the ERV.

Only the lid of the sample canister is exposed while on the Mars surface. Much of the contamination that clings to the lid can be expected to be removed during MAV ascent since it will receive the brunt of the aerodynamic heating and loading.

Contaminents that might be transferred to the docking cone will be eliminated with the jettisoning of the cone after the sample has been transferred.

One possible method for passivating contamination that might still be carried into the ERV on the canister lid would be a contact heating system to locally sterilize the lid surface.

SAMPLE TRANSFER AND CONTAMINATION CONTROL



EARTH ENTRY MODULE WITH SAMPLE CANISTER

The baseline MSSR mission for the purposes of this study assumes the returned sample will enter the Earth's atmosphere directly and be recovered by air snatch.

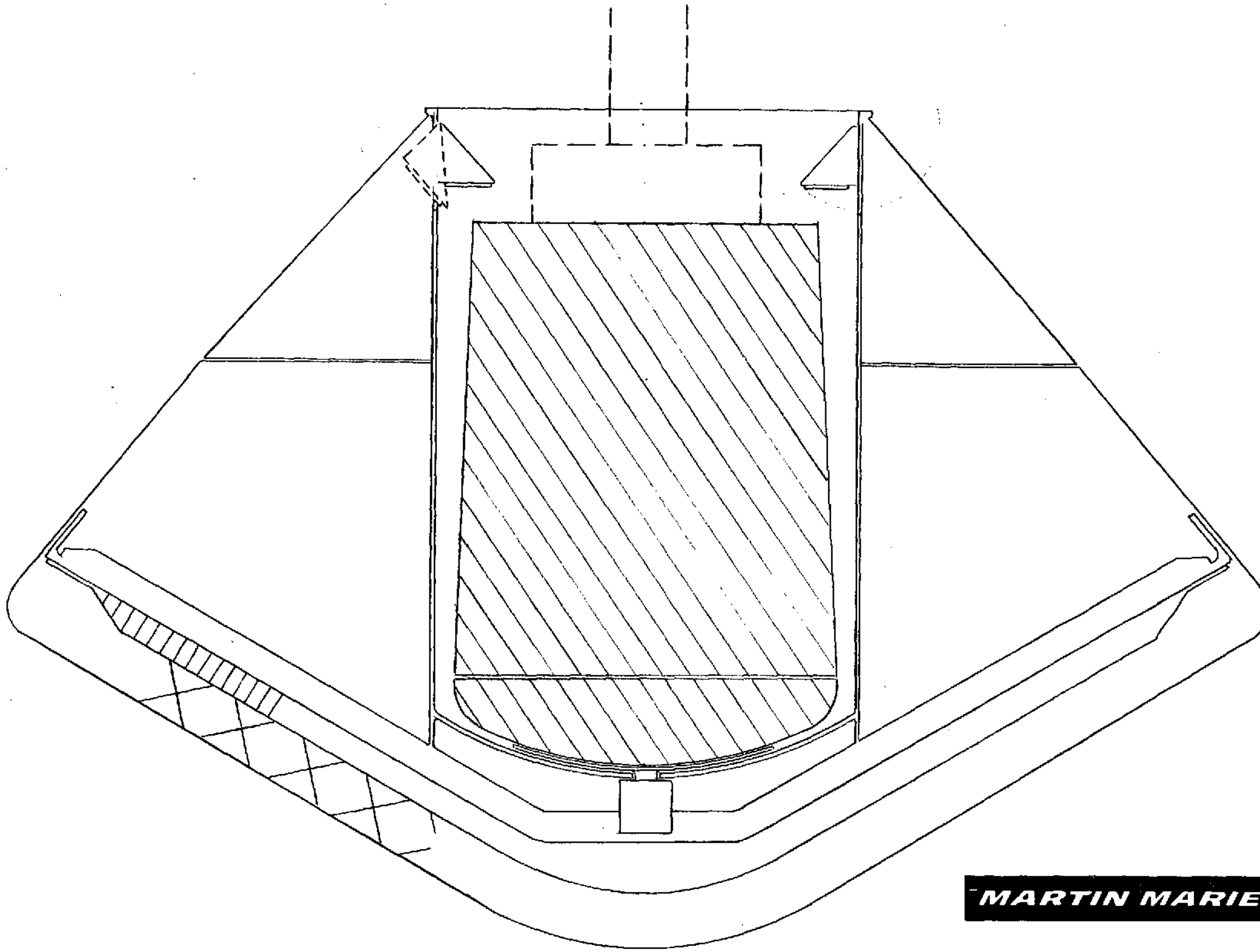
The Earth entry module shown here will mount in the ERV, receive the sample canister and finally be separated for Earth entry.

It contains a tracking beacon, parachute, heat shield and power subsystem.

The sample canister, after passing by the spring loaded trapping lugs and actuating the bottoming sensor, is driven back against the lugs to achieve a snug stowage condition within the entry module.

Weight allocation for the entry module is approximately 16 kg (35 lbs).

EARTH ENTRY MODULE WITH SAMPLE CANISTER



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SAMPLE CANISTER CONCEPT

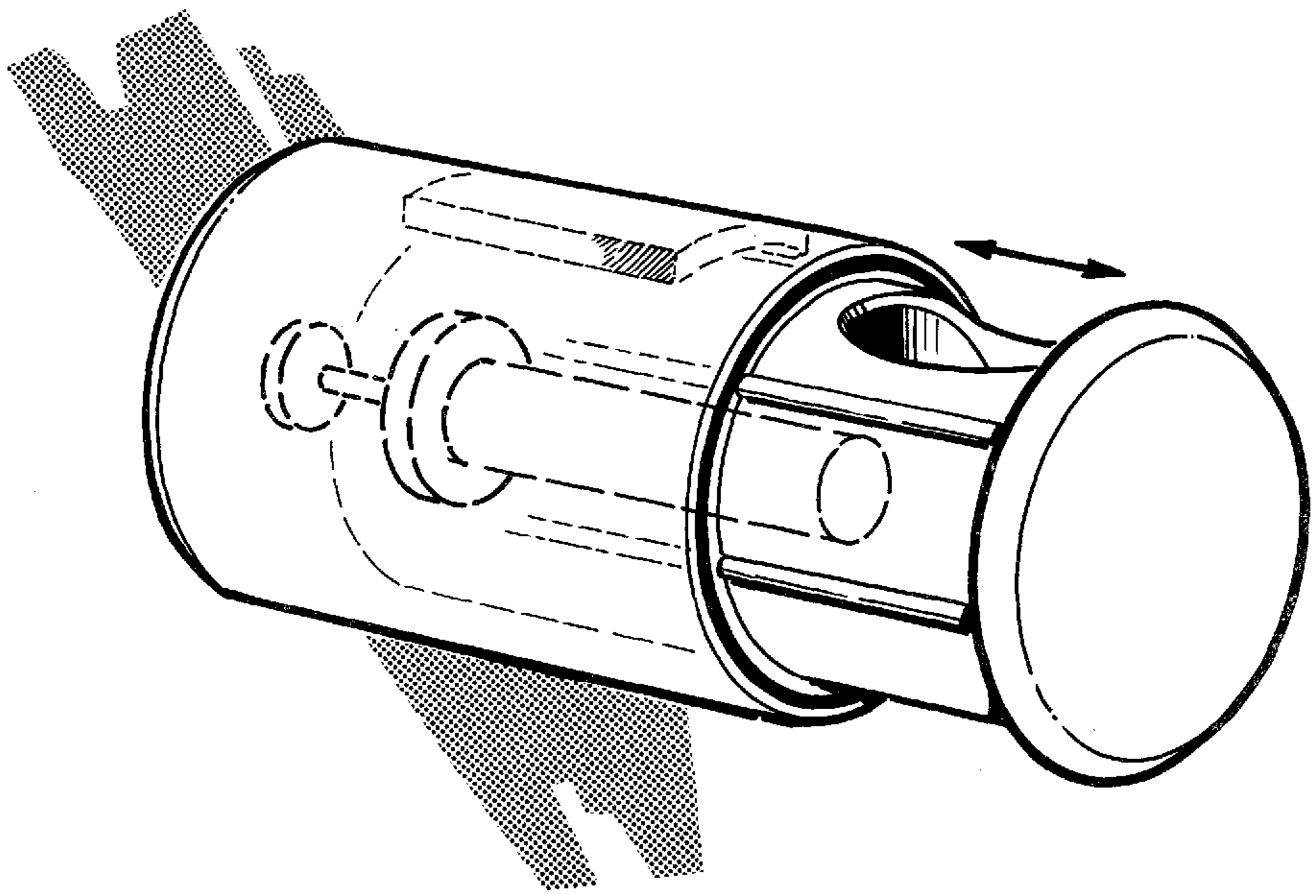
This concept uses a self contained actuator to extend the inner canister for sample loading and then draw it back and seat the seal.

Martin Marietta has been studying gold deforming seals of this type under contract to the Ames Research Center as part of an advanced Mars life detection experiment.

The baseline canister is designed to receive a bulk grab sample. Other concepts could receive capsules of sample taken from different locations that have been previously sealed by the sampling device.

The weight allocation for the sample canister is 0.91 kg (2 lbs).

SAMPLE CANISTER CONCEPT



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MSSR LAUNCH/ENCOUNTER SPACE

Twenty-day launch windows have been defined for the two Earth-Mars opportunities opening in 1981 and 1983/84. These windows have been optimized to maximize useful (non-propulsive) weight in a 2200 km circular Mars orbit, after subtracting a nominal weight allocation to the Lander/MAV configuration, which enters directly. That allocation has been sized for the 1981 mission at 1360 kg, providing a MAV liftoff weight of 288 kg. An additional 14 kg is allocated for the orbiter-lander adapter which is jettisoned prior to MOI, yielding a total cruise weight of 1374 kg not orbited.

The launch vehicle assumed is Titan IIIE/Centaur, and orbit insertion propulsion is Viking class. As currently configured, the MSSR design requires the minimum useful weight of 904 kg provided in 1981. For 1983/84, the lower orbited weights will necessitate fundamental changes to mission strategy.

MSSR LAUNCH/ENCOUNTER SPACE

1981 MSSR

Day	Launch	Arrival	C_3 $(km/sec)^2$	Injected Weight (kg)	Θ (deg)	Vhp (km/sec)	Useful Orbited* Weight (kg) to 2200 km
1	11-17-81	9-15-82	10.60	4185	221	3.06	907
10	11-26-81	9-21-82	9.41	4273	216	3.05	940
20	12-6-81	10-4-82	9.08	4244	213	3.15	904

1983/84 MSSR

Day	Launch	Arrival	C_3 $(km/sec)^2$	Injected Weight (kg)	Θ (deg)	Vhp (km/sec)	Useful Orbited* Weight (kg) to 2200 km
1	12-23-83	9-29-84	12.61	4045	221	3.53	739
10	1-1-84	10-7-84	11.35	4132	216	3.57	754
20	1-11-84	10-17-84	10.55	4189	213	3.69	739

*1374 kg associated with VLC/MAV enters directly

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MOI PROFILE

This orbit transfer sequence illustrates the MOI strategy proposed for the MSSR in 1981. The first impulse transfers the spacecraft to a "loose" capture orbit with a 1000 km periapsis, e = .9185, and orbital period of 105 hours. This orbit is held for 10-15 days while the Mars surface landing and sample acquisition takes place, followed by MAV ascent and establishment of the rendezvous orbit. At that time the final two orbit transfer maneuvers are performed. The second MOI burn raises periapsis to the MAV orbit altitude (2200 km nominally), and the third burn circularizes the orbit at periapsis.

MOI PROFILE

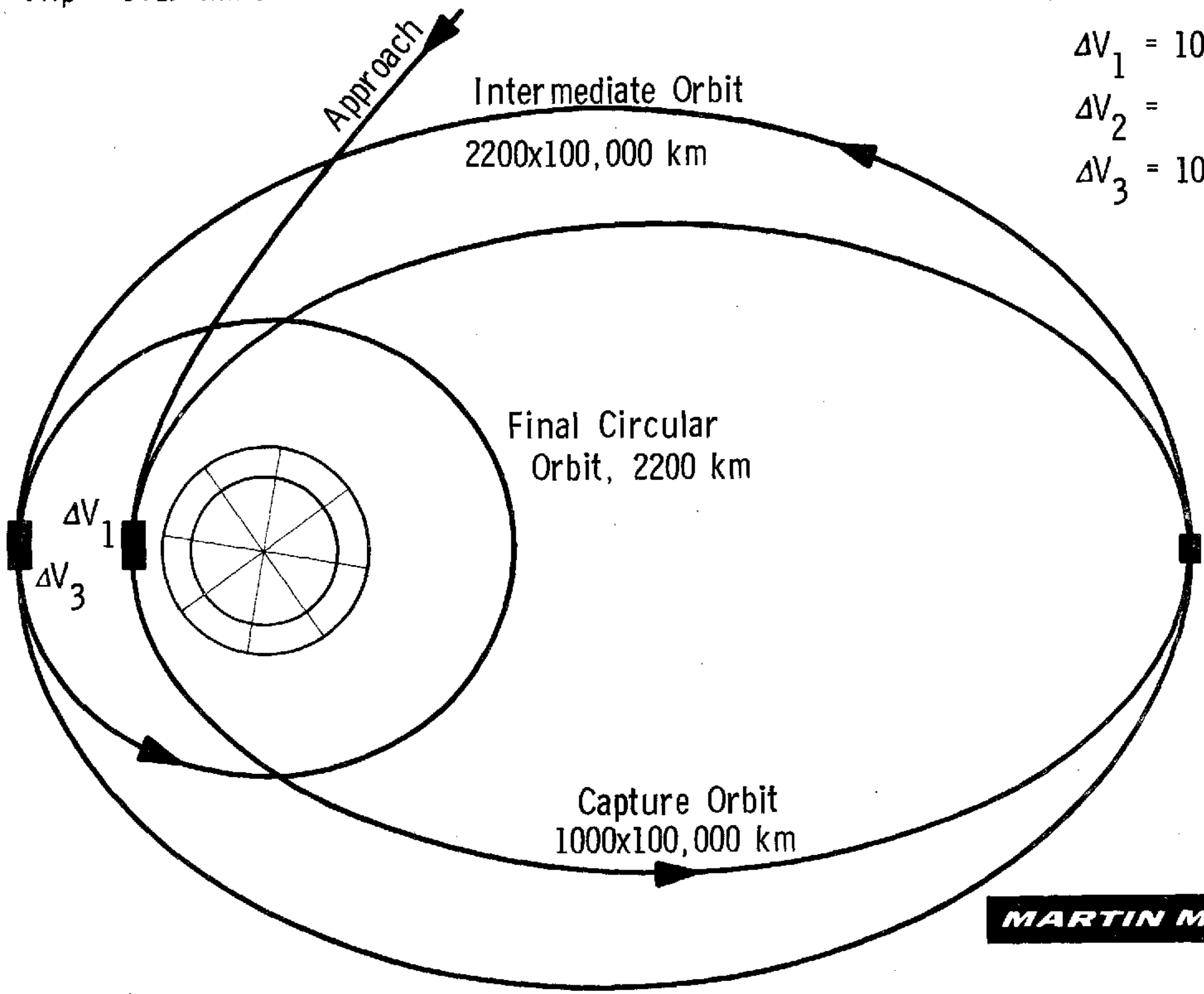
Vhp = 3.15 km/sec

Typical Insertion ΔV:

ΔV_1 = 1098 m/sec

ΔV_2 = 22 m/sec

ΔV_3 = 1044 m/sec



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TYPICAL ORBITER ΔV BUDGET

The ΔV capability provided the orbiter propulsion includes impulsive requirements to achieve the 3-impulse MOI to 2200 km circular (2.164 km/sec), plus an additional budget of .335 km/sec to account for midcourse corrections, finite burn losses, statistical ΔV, and rendezvous/trims.

TYPICAL ORBITER ΔV BUDGET

1981 Opportunity, 2200 km Circular Orbit, 20 Day Launch Window

Impulsive MOI (Vhp = 3.15 km/sec)			2.164 km/sec
ΔV_1 (1000 x 100,000 km)	=	1.098 km/sec	
ΔV_2 (2200 x 100,000 km)	=	0.022 km/sec	
ΔV_3 (2200 Circular)	=	1.044 km/sec	
Additional Budget			0.335 km/sec
MCC	=	0.035 km/sec	
Finite Burn Losses	=	0.100 km/sec	
ΔV_{stat}	=	0.050 km/sec	
Rendezvous and Trims	=	0.150 km/sec	

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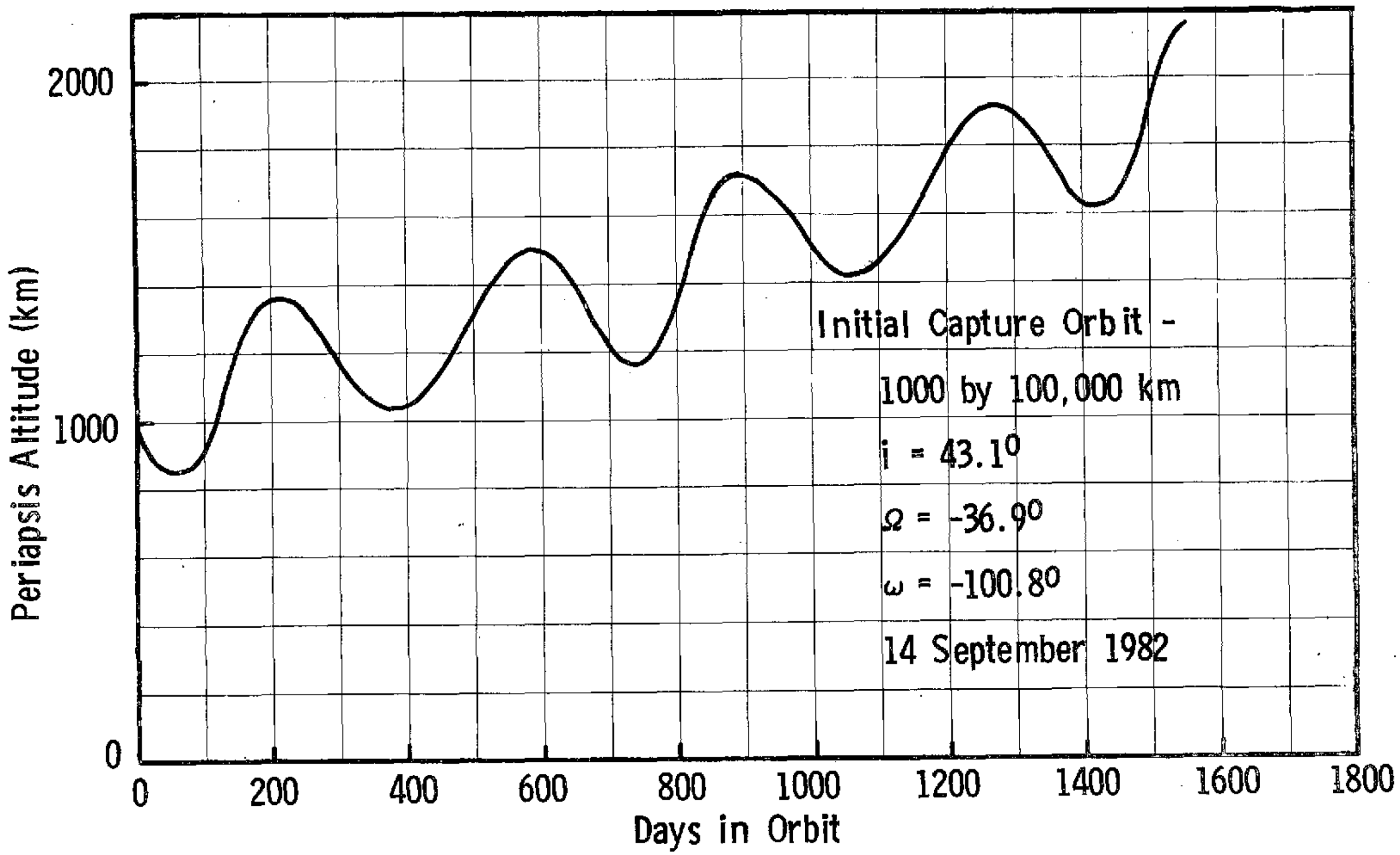
CAPTURE ORBIT STABILITY

With the proposed MSSR baseline, the orbit orientation (θ_{AIM}) has been selected to yield the unique inclination which contains both the incoming arrival asymptote and the departure Earth-return asymptote corresponding to a return window in November 1983. Orbital elements of that orbit are listed in the figure. Periapsis altitude stability for the 1000 by 100,000 km capture orbit at the desired orientation has been examined and found to exhibit an increasing character over the long term. This curve traces a 5-year history, and the trend continues for at least 50 years, considering a gravity model which includes solar perturbations and J2, ignoring Mars atmosphere at these altitudes.

CAPTURE ORBIT STABILITY



LANDED WEIGHT ASSUMPTIONS

For the analysis of lander performance in terms of what dry weights can be landed for various entry weights (at direct entry velocities), certain assumptions were made and are listed here. The mean Mars atmosphere is considered nominal, and landing is designed for mean surface level, or zero terrain height. L/D and parachute diameter are nominal Viking values. Entry conditions are sized by the maximum Vhp characteristic of each mission opportunity. Entry velocity is the velocity on the hyperbola at 800,000 feet altitude. Minimum entry angle is set $.5^{o}$ or more below the skipout angle for each entry velocity, representing the shallow end of the entry corridor. For the lander terminal descent propulsion, a pressure regulated system is assumed.

LANDED WEIGHT ASSUMPTIONS

- Mean Martian Atmosphere
- Zero Terrain Height (Landing at Mean Surface Level)
- L/D = 0.2 $\pm$ 0.02
- Parachute Diameter = 53 ft
- 1981: Max. $\begin{cases} V_{hp} = 3.15 \text{ km/sec} \\ V_E = 18981 \text{ fps} \end{cases}$ Min. $\gamma_E = -17.6^0$
- 1983/84: Max. $\begin{cases} V_{hp} = 3.70 \text{ km/sec} \\ V_E = 20021 \text{ fps} \end{cases}$ Min. $\gamma_E = -18.1^0$
- Pressure Regulated Terminal Descent Propulsion

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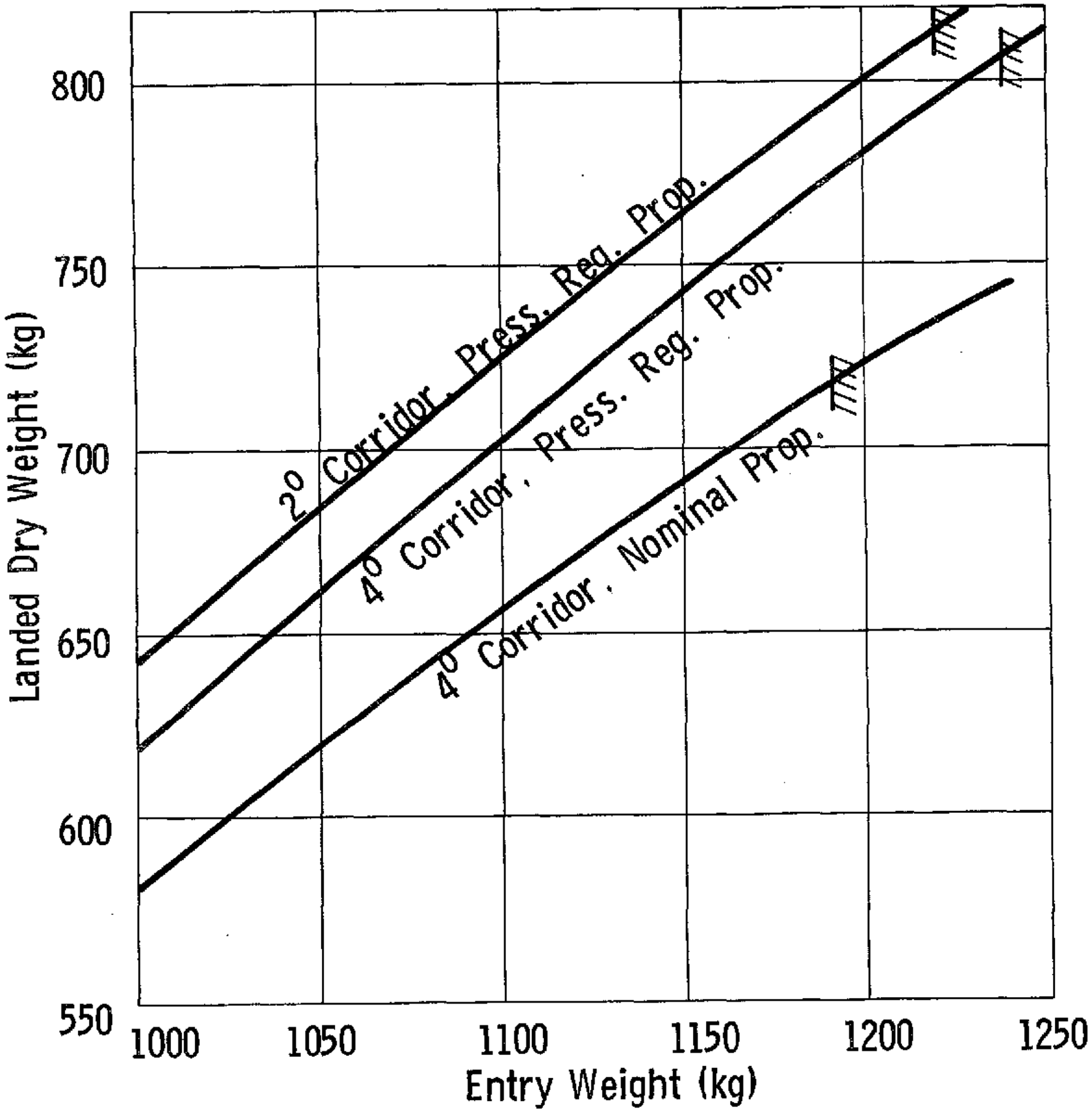
LANDED WEIGHT CAPABILITY

Entry corridor widths of 2^{o} and 4^{o} are compared in this figure, with landed weight capability the measure of performance. The upper curves both assume pressure regulated terminal descent propulsion, differing only in width of corridor. With the wider corridor, steeper descent conditions necessitate a heavier aeroshell, and cut 20 kg from landed dry weight potential. Comparison of the two lower curves, both for 4^{o} corridor widths, shows the significant performance enhancement gained by the pressure regulated system - between 40 to 60 kg in landed weight.

Current studies indicate the direct entry mode for MSSR would require optical navigation to ensure a 2^{o} entry corridor, while DSN tracking is sufficient with the 4^{o} corridor. The wider corridor, with pressure regulated propulsion, has therefore been selected as our reference.

LANDED WEIGHT CAPABILITY



Assumes:

- Mean Mars Atmosphere
- Zero Terrain Height
- Vhp = 3.15 km/sec
- L/D = 0.2 ± 0.02

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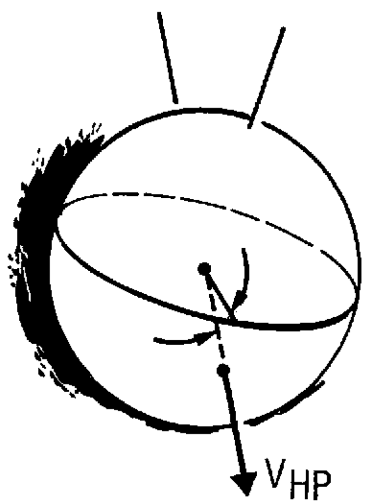
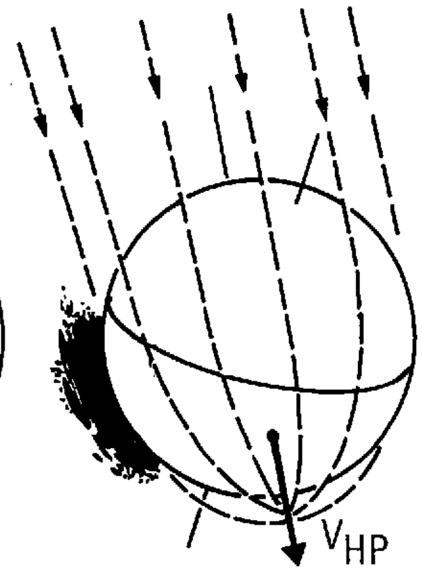
LANDING SITE RESTRICTIONS FOR DIRECT ENTRY LANDERS

This series of illustrations indicates the generalized constraints on landing sites for landing trajectories from the incoming asymptote. For the baseline 1981 mission used in this study the inclination of the Vhp vector is approximately -30° (to the Mars equatorial plane). A number of constraints actually apply to the final landing latitude accessibility deriving from communications, navigation, Sun elevation angle requirements, etc., but generally speaking landing sites in the Southern hemisphere will be favored.

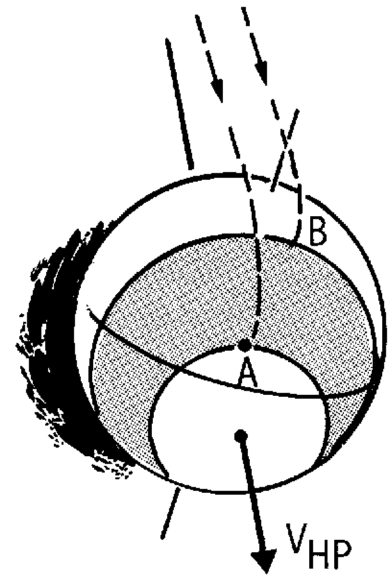
LANDING SITE RESTRICTIONS FOR DIRECT ENTRY LANDERS



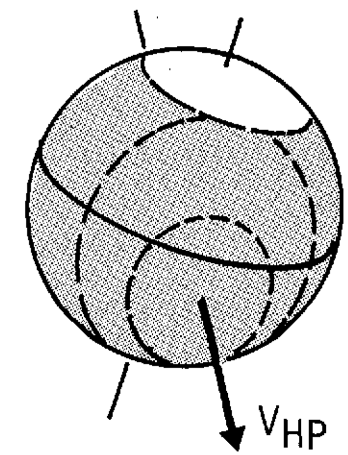
Inclination Of Incoming Asymptote Determined By Planetary Geometry (Mission Year, Etc.)



Possible Incoming Lander Trajectories Are Symmetrical About The V_{HP} (Selectable By Appropriate Midcourse Correction)



Accessible Landing Area Is Restricted By: (A) Entry Too Shallow (Skip-out); and (B) Entry Too Steep (Parachute Opening Mach No. Too High)

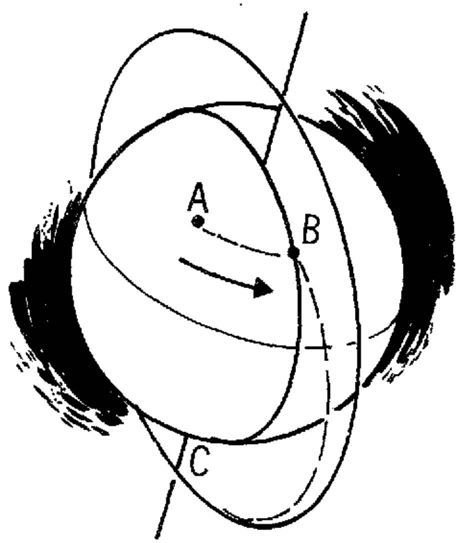


Accessible Landing Latitudes Can Generally Be Achieved At Any Longitude By Adjusting The Arrival Time At Mars Of Each Lander

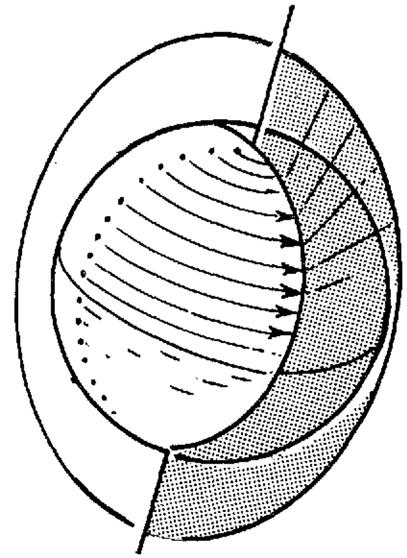
LANDING SITE RESTRICTIONS DUE TO RENDEZVOUS ORBIT

For a minimum performance mission, the landing site must pass under the orbit plane during the planet rotation. This illustration shows how this constraint will affect the landing latitude accessibility. For this baseline 1981 mission, given navigation constraints and the requirement that the rendezvous orbit contain, as nearly as possible, the incoming and outgoing Vhp vectors, the landing sites will be restricted to the near equatorial regions.

LANDING SITE RESTRICTIONS DUE TO RENDEZVOUS ORBIT



Acceptable Landing Sites (A) Should Rotate Into Plane Of Rendezvous Orbit For Launch (B) To Rendezvous (C). Otherwise Costly Plane Changes Are Required.



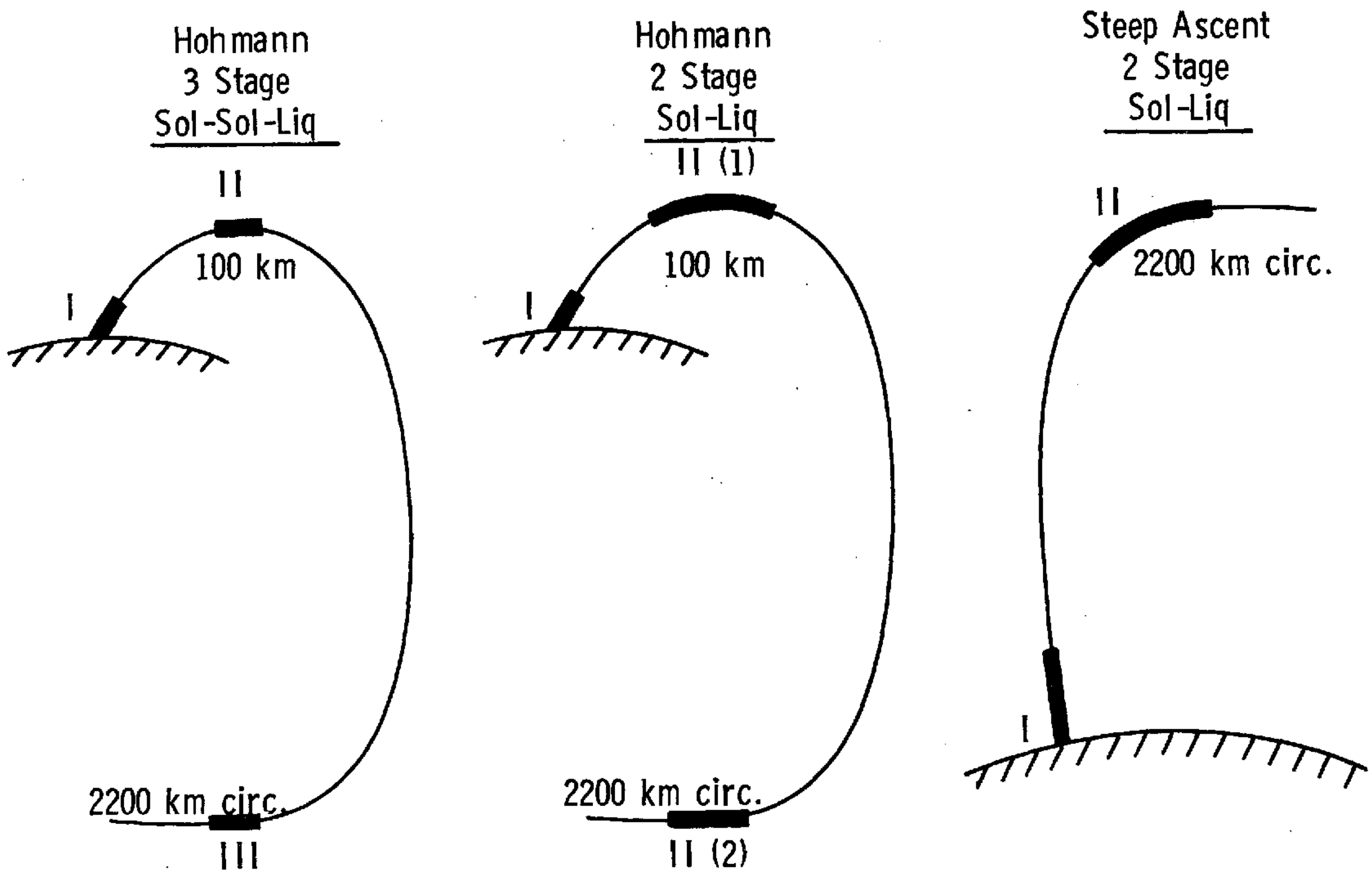
Polar Orbits (90^0 Inclination) Can Pick Up Landers From Any Latitude. However, MAV Launches To Equatorial Inclinations Are Easier Than Polar Inclinations

North And South Latitudes Above The Inclination Of The Rendezvous Orbit Would Not Be Good Landing Areas.

MAV ASCENT PROFILES

Three proposed MAV ascent profiles have been examined to assess their potential for delivering a sample payload into a circular rendezvous orbit. The pictorial on the left illustrates a 3-stage sequence involving 1) a solid stage boost to 100 km, 2) a second solid stage burn to an elliptic orbit, and 3) a liquid third stage circularization burn. In the center pictorial the trajectory is similar, but the second and third ascent burns are performed by a single liquid stage. The profile on the right involves a solid stage "steep ascent" directly to the final rendezvous orbit altitude, followed by a second liquid stage burn which circularizes at that point.

MAV ASCENT PROFILES



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ASCENT PROFILE COMPARISON

This table summarizes the performance aspects of the three proposed ascent profiles for the MAV. For the comparison, a typical MAV weight of 250 kg is assumed, with rendezvous in a 2200 km circular orbit. After optimization of staging for each case, final stage non-propulsive weights are compared. The results indicate the 3-stage solid-solid-liquid profile to be the most efficient strategy for delivering the Mars sample to circular rendezvous orbit.

ASCENT PROFILE COMPARISON . . . 250 kg MAV to 2200 km

		Hohmann 3 Stage Sol-Sol-Liq	Hohmann 2 Stage Sol-Liq	Steep Ascent 2 Stage Sol-Liq
Stage I	Isp (sec)	285	285	285
	Mass Fraction	.88	.88	.88
	Weight (kg)	128	126	207
Stage II	Isp (sec)	285	295	295
	Mass Fraction	.88	.70	.70
	Weight (kg)	88	124	43
Stage III	Isp (sec)	235	None	None
	Mass Fraction	.40		
	Weight (kg)	34		
Final Stage Non-propulsive Weight (kg)		18	7	8

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1981 MISSION WEIGHT ALLOCATION TRADE

This table presents end-of-window weight possibilities for the 1981 opportunity, with a launch window length of 20 days. After defining weight requirements for the basic mission spacecraft components, a performance trade exists in the distribution of remaining weight between orbiter propulsion and the Lander/MAV configuration. Allocating more weight to the Lander/MAV translates into a larger, heavier MAV, which gains more final stage payload, or the potential to reach a higher rendezvous orbit altitude. If instead weight allocations are directed toward a larger orbiter propulsion system, the orbiter can gain a lower circular rendezvous orbit, thereby easing the requirements on MAV stage propulsion, leading to smaller, lighter MAV designs.

1981 MISSION WEIGHT ALLOCATION TRADE (20 Day Launch Window)

Launch Weight	4409 kg
Adapter and LVMP	165
Injected Weight. Cruise	4244
Spacecraft at MOI	
Orbiter Bus	600
Earth Return Vehicle	263
VPR	41
Propellant	1692
Inerts	274
Lander/Mav Configuration	1360
Orbiter-Lander Adapter	14

$\sum = 3326$

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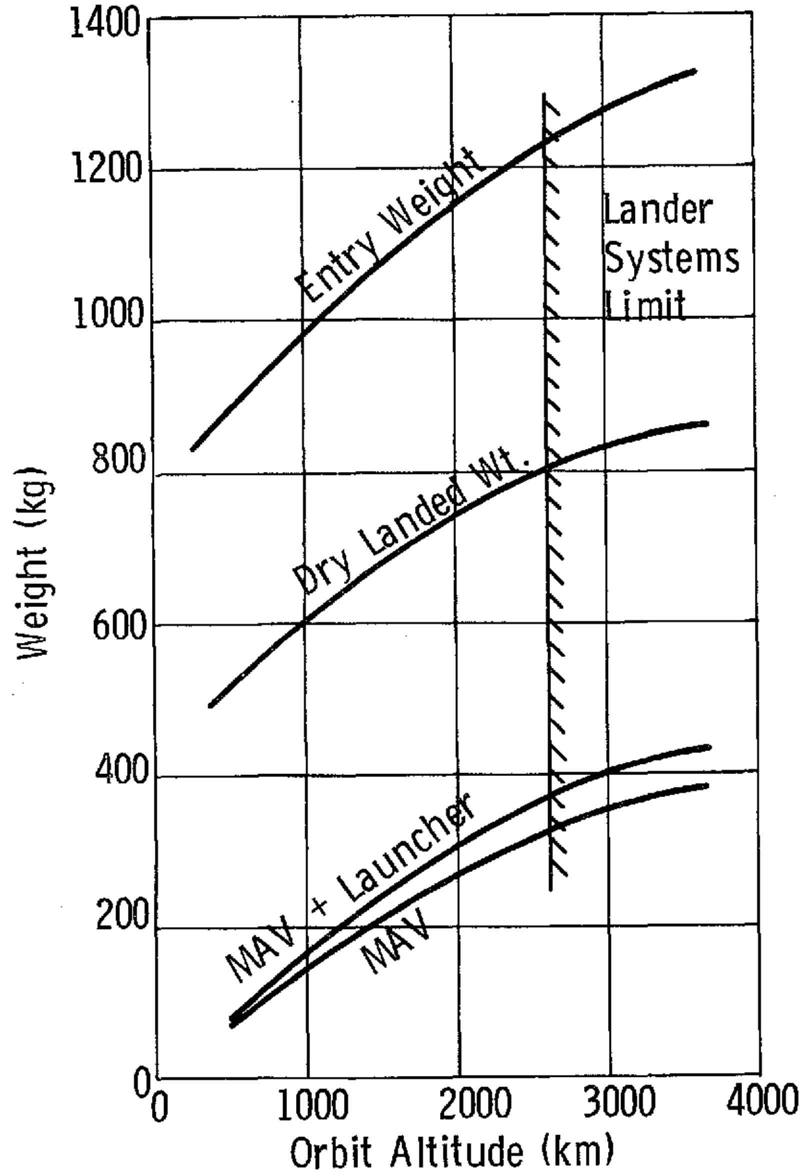
LANDED WEIGHT VS RENDEZVOUS ORBIT ALTITUDE

These curves illustrate the effect of trading weight between Lander/MAV and orbiter propulsion. The entry weight curve is a direct measure of the cost in higher rendezvous orbit altitudes as Lander/ MAV weight increases. The lower curves translate entry weight into dry landed weight, and finally into weight available for the MAV itself, as orbit altitude varies with orbiter propulsion weight. (12% of the MAV + lander weight is assigned to the lander mechanism.)

LANDED WEIGHT VS RENDEZVOUS ORBIT ALTITUDE



Assumes:

- 1981, 20 Day Window
- 4^{0} Entry Corridor
- Press. Reg. Lander Prop.
- 435 kg Basic Lander

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MAV FINAL STAGE WEIGHT SENSITIVITIES

To gain understanding of the mission trade involving weight distribution between orbiter propulsion and Lander/MAV, that is, lower rendezvous orbits versus heavier MAV weights, the sentitivity of final stage MAV non-propulsive weight to those parameters was determined. That weight, defined here as P/L, provides a quantitative measure of ultimate mission performance - what can be delivered from the Mars surface to an orbital rendezvous. The sensitivities defined allow the evaluation of each combination of Lander/MAV weight and orbit altitude in terms of P/L, and thus provide a method for optimizing mission performance. Again, it should be noted that here P/L refers to <u>all</u> stage III non-propulsive weight, not the surface sample alone.

MAV FINAL STAGE WEIGHT SENSITIVITIES

- Hohmann Ascent Profile, 3 Stage MAV, Sol-Sol-Liquid
- Theoretical Stage III Mass Fraction = 0.4
- P/L ≡ All Non-propulsive Stage III
- Reference: 288 kg MAV to 2200 km, P/L = 24.6 kg

$$\frac{\partial \text{ P/L}}{\partial \text{ MAV (Liftoff)}} = +0.0715 \text{ kg/kg}$$

$$\frac{\partial \text{ P/L}}{\partial \text{ Rend. Orbit Altitude}} = -0.0047 \text{ kg/km}$$

+14 kg MAV (Liftoff) ⟶ +1 kg P/L ⟵ -213 km Orbit Alt.

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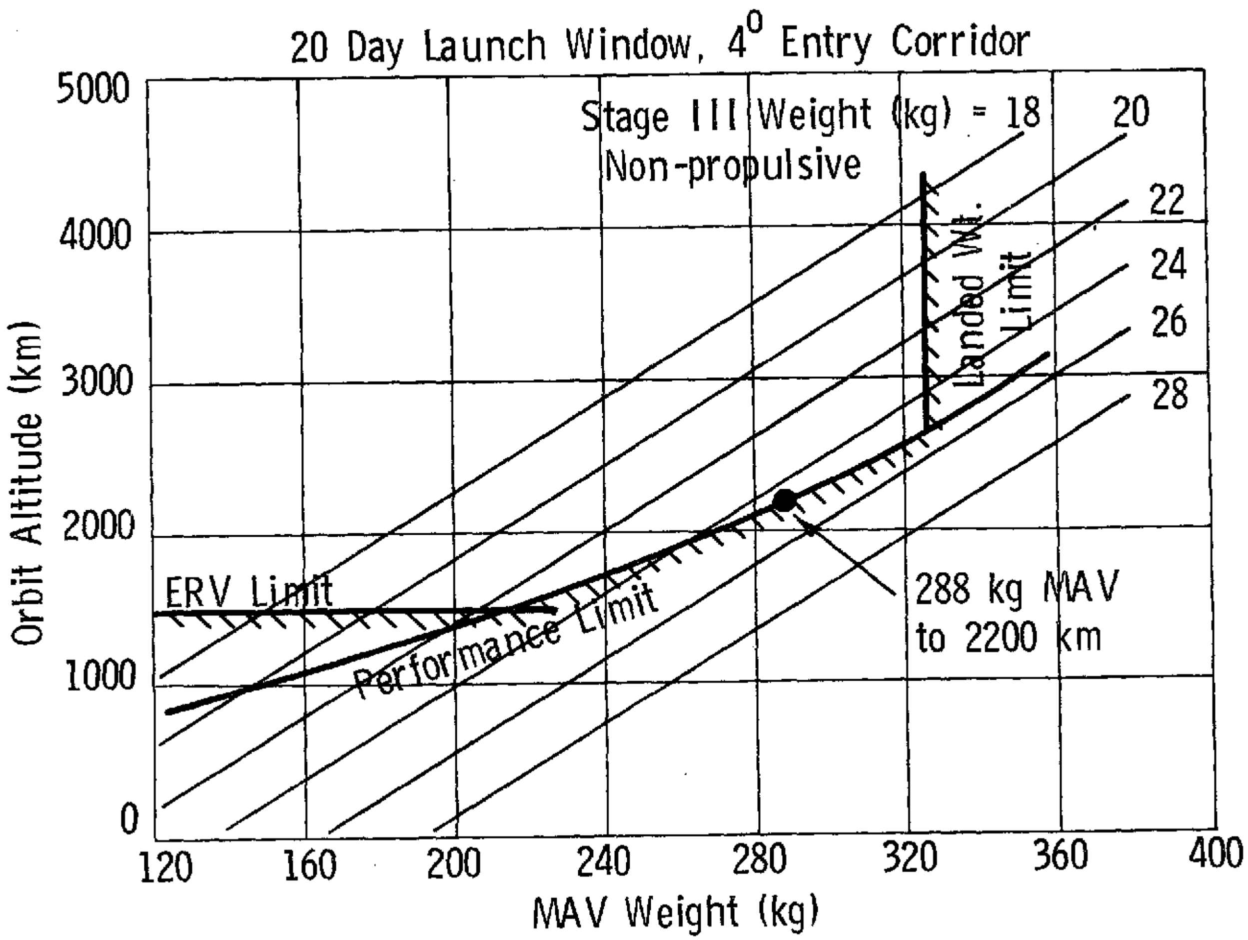
MAV PERFORMANCE DESIGN TRADE FOR 1981

This figure presents the critical performance design curves and constraints which apply to the 1981 MSSR as currently configured - assuming a 20-day launch window and 4° entry corridor. The curve labeled "performance limit" represents what is achievable, given baseline weight allocations and system performance. All points above the curve are theoretically possible. Points <u>on</u> the curve indicate use of full mission capability. The "ERV limit" defines the lowest circular orbit from which the Earth Return Vehicle can achieve transfer to the return trajectory. "Landed weight limit" derives from the heaviest entry weight which the lander system can handle.

Superimposed over the curves are lines of constant MAV stage III non-propulsive weight, which are approximated from the sensitivity analysis. The design trade indicates an optimum P/L near 25 kg, constrained by the landed weight limit to a MAV weight of 325 kg at 2600 km orbit altitude. Due to configuration problems associated with containment of a large MAV within the lander, the proposed baseline is backed-off to 288 kg at 2200 km rendezvous orbit altitude. Relative flatness of the performance curve with respect to P/L contours in this region yields only a small sacrifice in stage III non-propulsive weight, reduced to about 24.4 kg.

MAV PERFORMANCE DESIGN TRADE FOR 1981



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PROPOSED ORBITER/LANDER/MAV WEIGHTS FOR 1981

This list summarizes the proposed weight allocations to various MSSR spacecraft modules at primary phases of the mission. The baseline MAV design is 288 kg, with a rendezvous orbit altitude of 2200 km. A 20-day launch window is assumed. A Mars direct entry corridor width of 4° is considered, with pressure regulated terminal propulsion for the lander.

PROPOSED ORBITER/LANDER/MAV WEIGHTS FOR 1981

20 Day Launch Window, 4^0 Entry Corridor, 2200 km Rendezvous Orbit

Launch Weight	4409 kg
Injected Weight, Cruise	4244
Spacecraft at MOI	2870
Orbiter Bus	600
Earth Return Vehicle	263
VPR	41
Propellant	1692
Inerts	274
Orbiter-Lander Adapter	14

Lander/MAV Loaded	1360 kg
Weight After Separation	1249
Usable Deorbit Prop.	72
Entry Weight	1177
Dry Landed Weight	763
Basic Lander	435
MAV + Launcher	328
MAV	288

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MAV STAGING PHILOSOPHY

This list presents a weight and performance breakdown of the optimized staging for a 288 kg MAV to 2200 km circular orbit. For the third stage ΔV budget, an additional 50 m/sec is allocated for statistical ΔV and trims.

MAV STAGING OPTIMIZATION

288 kg MAV to 2200 km Rendezvous Orbit

Stage III	Non-propulsive Weight	= 24.4 kg	Isp = 235 sec
	Propellant	= 6.2	ΔV_{BUDG} = 391 m/sec
	Prop. Inerts	= 9.3	Mf = 0.40
	Total Weight	= 39.9	
Stage II	Skirt II-III	= 4.1	Isp = 285 sec
	Propellant	= 81.1	ΔV_{BUDG} = 2530 m/sec
	Prop. Inerts	= 11.1	Mf = 0.88
	Total Weight	= 96.3	
Stage I	Skirt I-II	= 5.7	Isp = 285 sec
	Propellant	= 128.6	ΔV_{BUDG} = 1654 m/sec
	Prop. Inerts	= 17.6	Mf = 0.88
	Total Weight	= 151.9	

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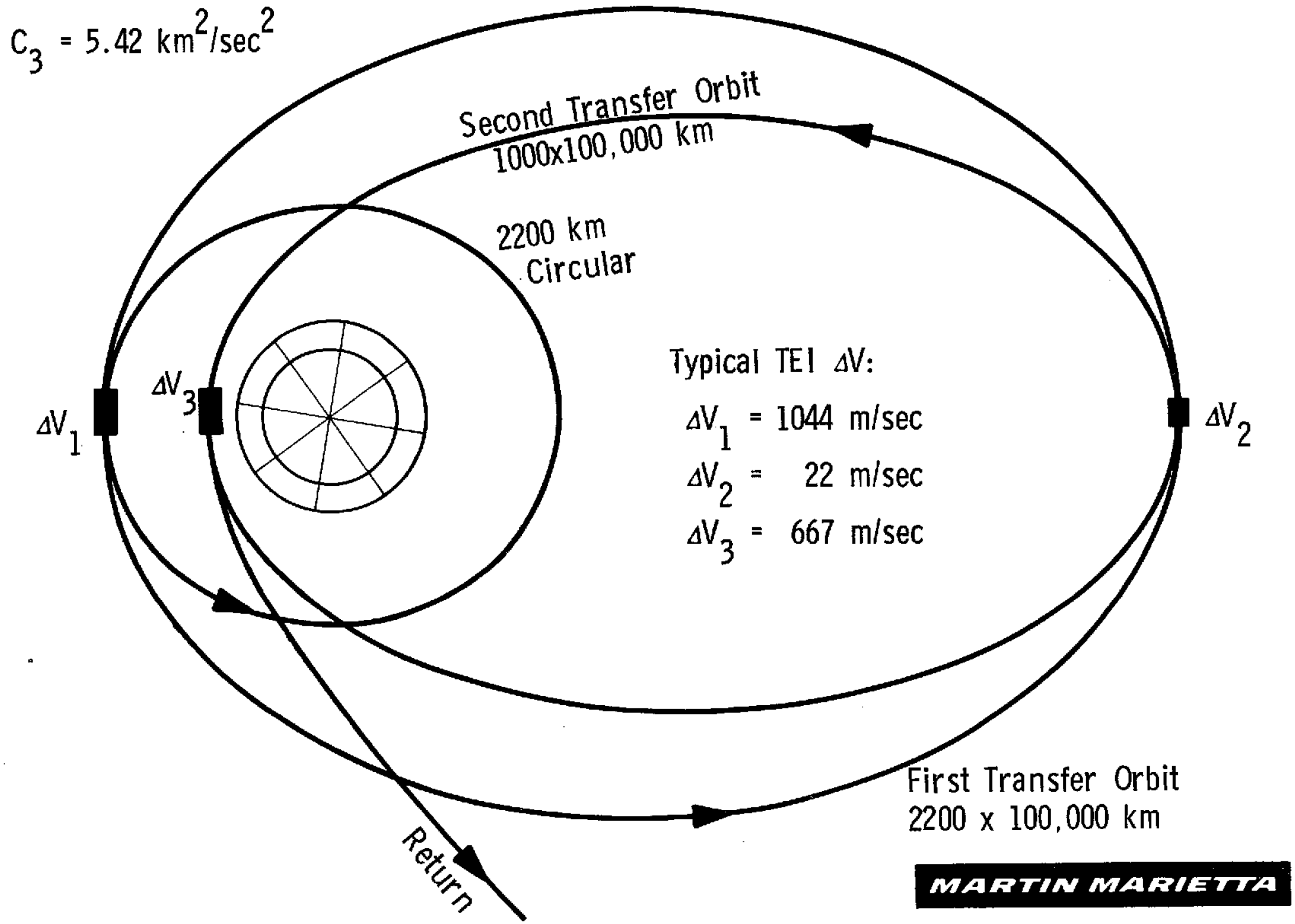
TEI PROFILE

The orbital transfer for the Earth-return trajectory is basically the reverse of MOI. Total impulsive ΔV is less than MOI since hyperbolic velocity for departure is less than that at arrival (2.33 versus 3.15 km/sec).

TEI (EARTH RETURN) PROFILE



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NAVIGATION ANALYSIS

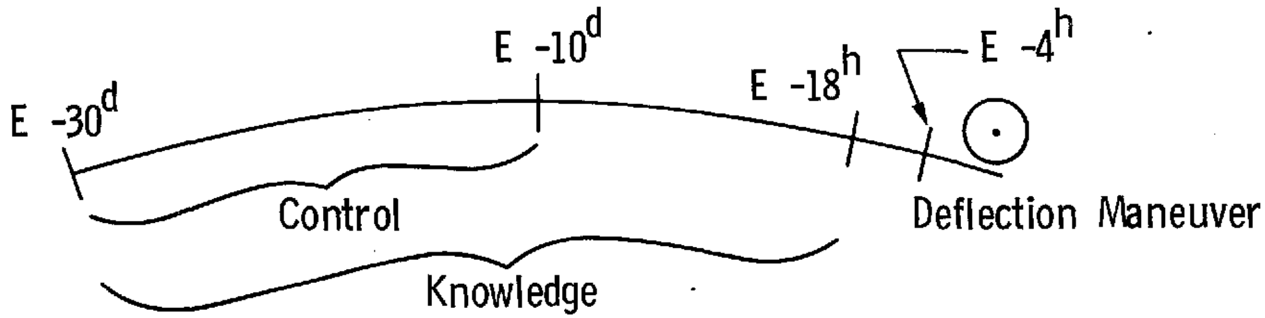
A. L. Satin

APPROACH GEOMETRY

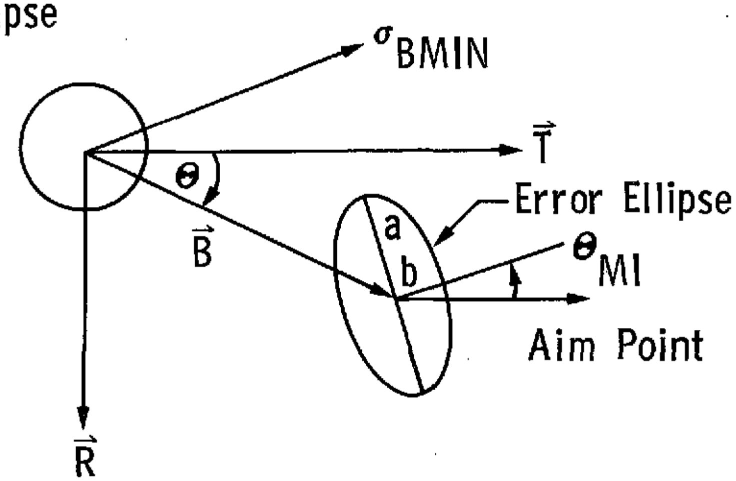
The approach tracking periods and deflection maneuver time is shown. Tracking data from E-30d to E-10d is used to target the last midcourse correction at E-10d. The orbit determination (O.D.) accuracy at this time limits the orbit "control" capability for deflection and MOI maneuvers. Tracking data for determination of the deflection maneuver is taken from E-30d to E-18^h.

State accuracies at this time limit represent the "knowledge" available to target deflection. Tracking down to E-12^h may be used to target the MOI maneuver. Statistics of state dispersions are represented by the B-plane error ellipse centered at the nominal B (impact) vector. The orientation of this ellipse is specified by the angle θ_{MI}. Note that the smallest dispersions in the B-vector magnitude occur when the B-vector is oriented along the ellipse minor axis (b).

Compute Control, Knowledge Matrices



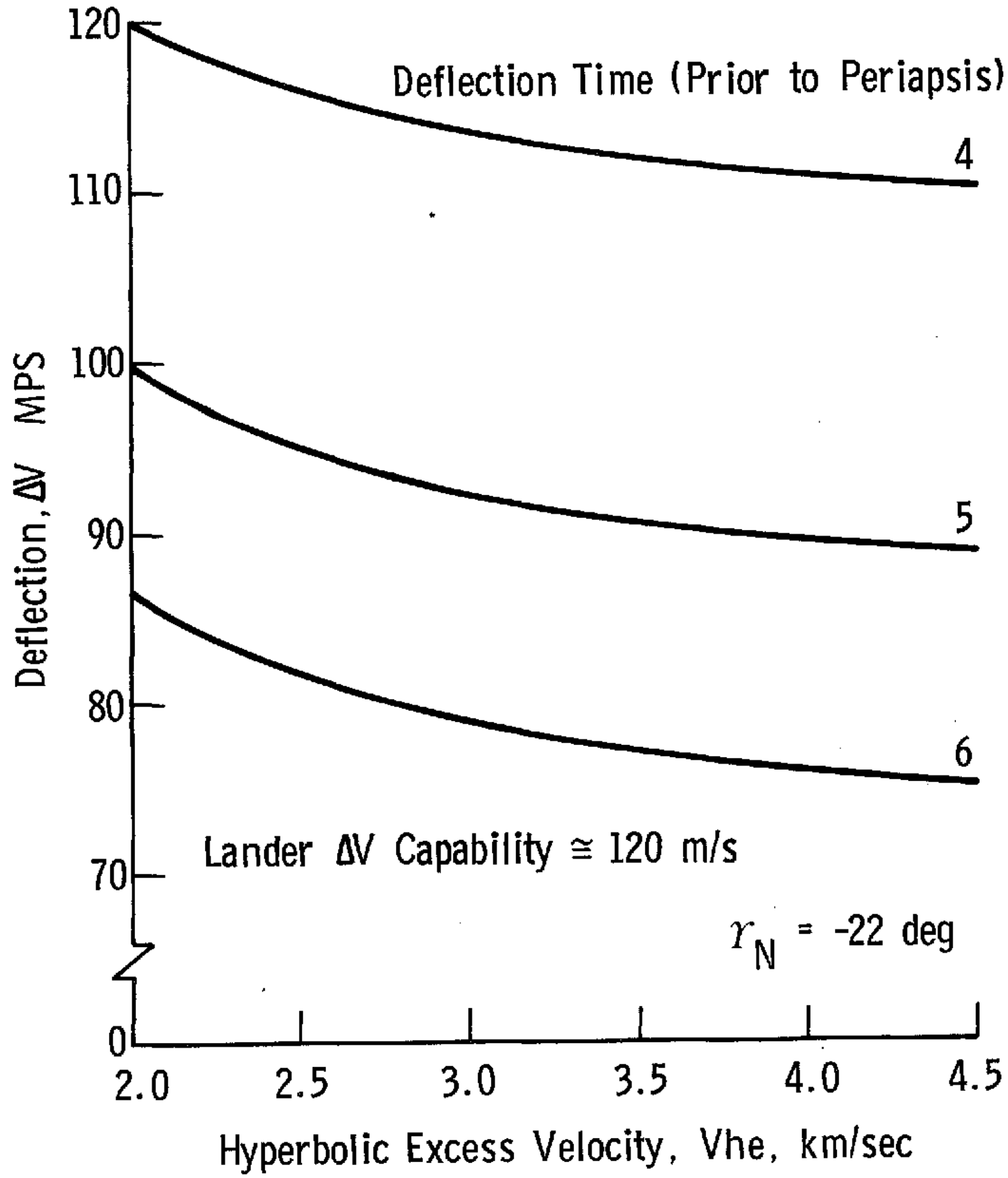
B Plane Error Ellipse



DEFLECTION MANEUVER ΔV REQUIREMENTS

A deflection maneuver 4 hours from encounter is affordable with the higher VHEs typical of the 1981 and 1983/84 MSSR missions. This is because the spacecraft is further from the planet at the fixed time for higher encounter velocities.

DEFLECTION MANEUVER ΔV REQUIREMENTS



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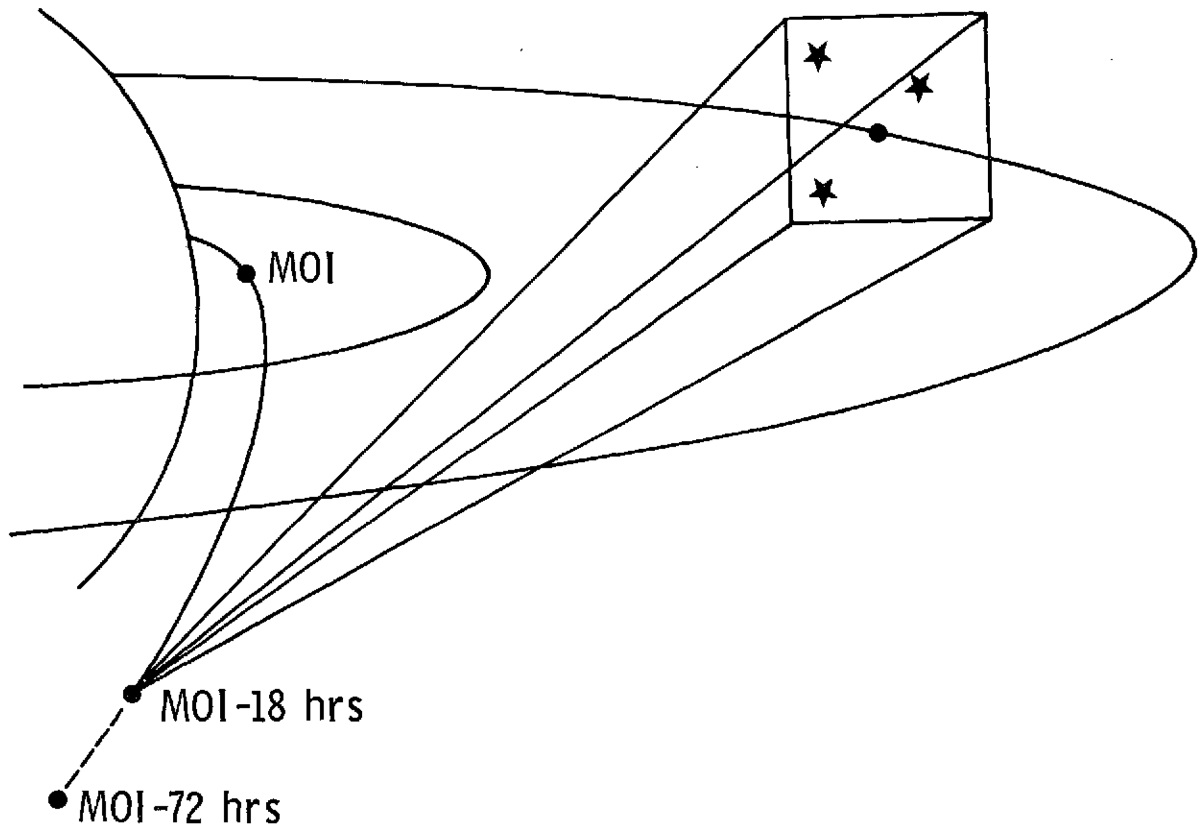
"KNOWLEDGE" DATA TYPE DEIMOS/STARS - SINGLE CAMERA

On board TV sightings of Deimos against a star background may be used to simultaneously solve for the spacecraft and satellite states. These sightings are taken from MOI-72 hrs to MOI-18 hrs. Typical B-plane ellipse major axes for this type of data are of the order of 25 km. This allows very accurate entry flight path control for any θ approach angle. Since a Mariner TV system weighs at least 30 lbs it was necessary to examine the tradeoff between corridor reduction and increased orbiter weight.

"KNOWLEDGE" DATA TYPE DEIMOS/STARS - SINGLE CAMERA



APPROACH OD: DSN VS DSN + OPTICAL

A comparison of entry corridor width ($6\sigma_\gamma$) with and without optical (TV) tracking is made. Radio only $\sigma_{/B/}$ capability is ~25 km assuming a Mars ephemerus error of the same magnitude. Radio + optical allows a 2° corridor width for any hyperbolic approach angle while radio-only affords 4° accuracy for a very restrictive approach angle (namely along the minor axis of the B-ellipse). A restrictive approach angle also means limited latitude accessibility.

APPROACH OD: DSN VS DSN + OPTICAL

Assumptions:

1) VHP = 3.15
   $\gamma = -18.5^0$
   RE = 3637.24

2) Radio: E-30d E-12 hrs
   Optical: E-3d E-18 hrs } Deimos

3) At least 1 star in satellite background
   Data noise only = 1 pixel

4) $$\sigma_\gamma = \frac{-V_\infty \ CSC\ \gamma}{R_E \sqrt{V_\infty^2 + 2\mu/R_E}} \sigma_{|B|}$$

Results:

	Radio Only			Radio + Optical
$\sigma_{\lvert B\rvert}$	25 km	50 km	75 km	12 km
$1\sigma_\gamma$	$.676^0$	1.352^0	2.027^0	.324
$6\sigma_\gamma$	4.056^0	8.112^0	12.162^0	1.946

Conclusions:

1) Optical (TV) Sightings Required to Achieve 2^0 Corridor

2) 4^0 Attainable with DSN (optional QVLBI)

3) Results with Optical Independent of LD/ED

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MISSION SCHEMATIC #1 - ORBITER CAPTURE TO MAV ASCENT

1) Orbiter performs MOI to loose capture orbit (1000 x 100,000 km). ΔV_{MOI} = 1098. Lander touches down near periapsis.

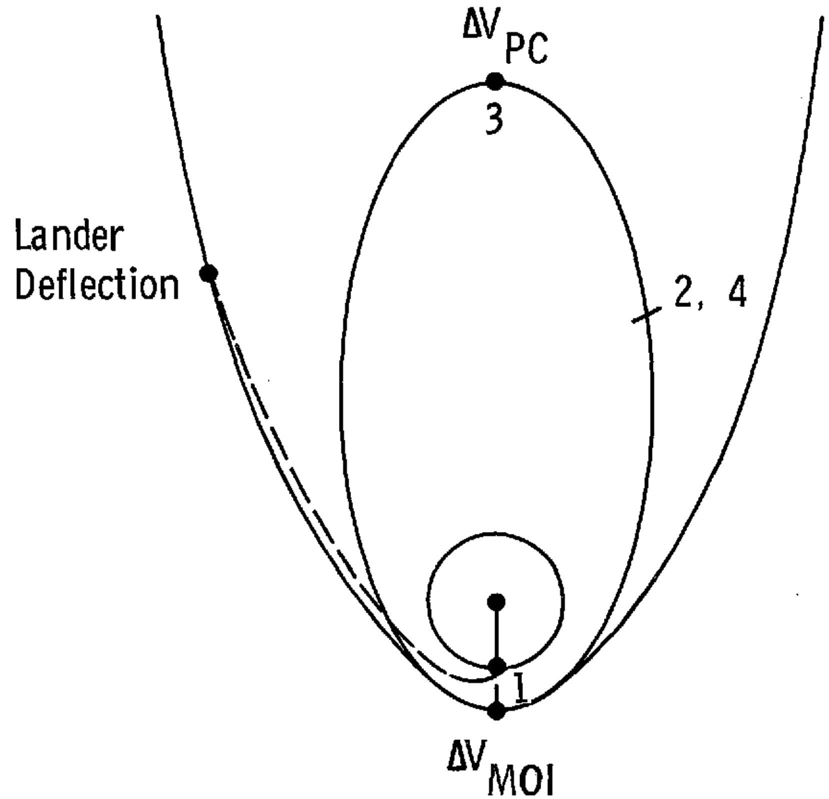
2) Orbiter state vector update based on ~1½ orbits of conventional DSN Doppler data.

3) ΔV_{PC} = orbiter plane change maneuver for return.

4) Final determination of orientation of orbiter plane of motion prior to MAV liftoff. Based on ~1 orbit of conventional DSN Doppler data.

5) MAV liftoff when orbiter at 3rd apoapsis.

MISSION SCHEMATIC #1 - ORBITER CAPTURE TO MAV ASCENT



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MISSION SCHEMATIC #2 - MAV ASCENT TO CIRCULARIZATION TRIM

6) MAV injection to 100 x 2200 km orbit

7) State vector update based on ~8 orbits of conventional Doppler

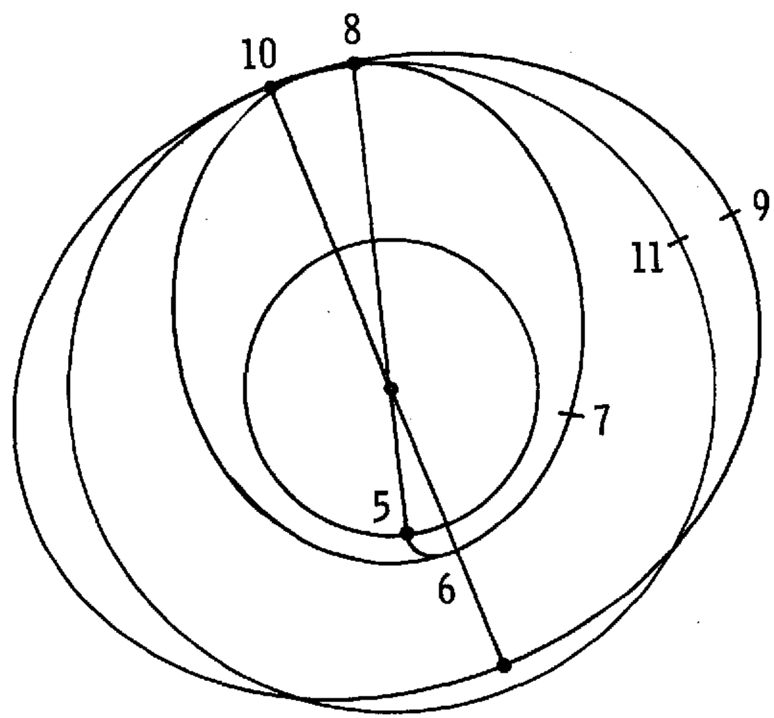
8) Circularization burn ($\Delta V \cong 306$ m/s)

9) State vector update based on 4 orbits of conventional Doppler

10) Circularization trim ($\Delta V = 0$)

11) State vector update based on ~4 orbits of conventional Doppler

MISSION SCHEMATIC #2 - MAV ASCENT TO CIRCULARIZATION TRIM



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MISSION SCHEMATIC #3 - ORBITER PERIAPSIS CHANGE TO ORBITER CIRCULATION TO 1^{st} OCCULTATION EXIT

12) State update based on 1 orbit of conventional doppler

13) Orbiter raises periapsis on 4^{th} apoapsis (ΔV = 26 m/s)

14) State vector update based on ~1 1/2 orbits of conventional doppler

15) Orbiter intermediate phasing burn (ΔV - 993.7 m/s)

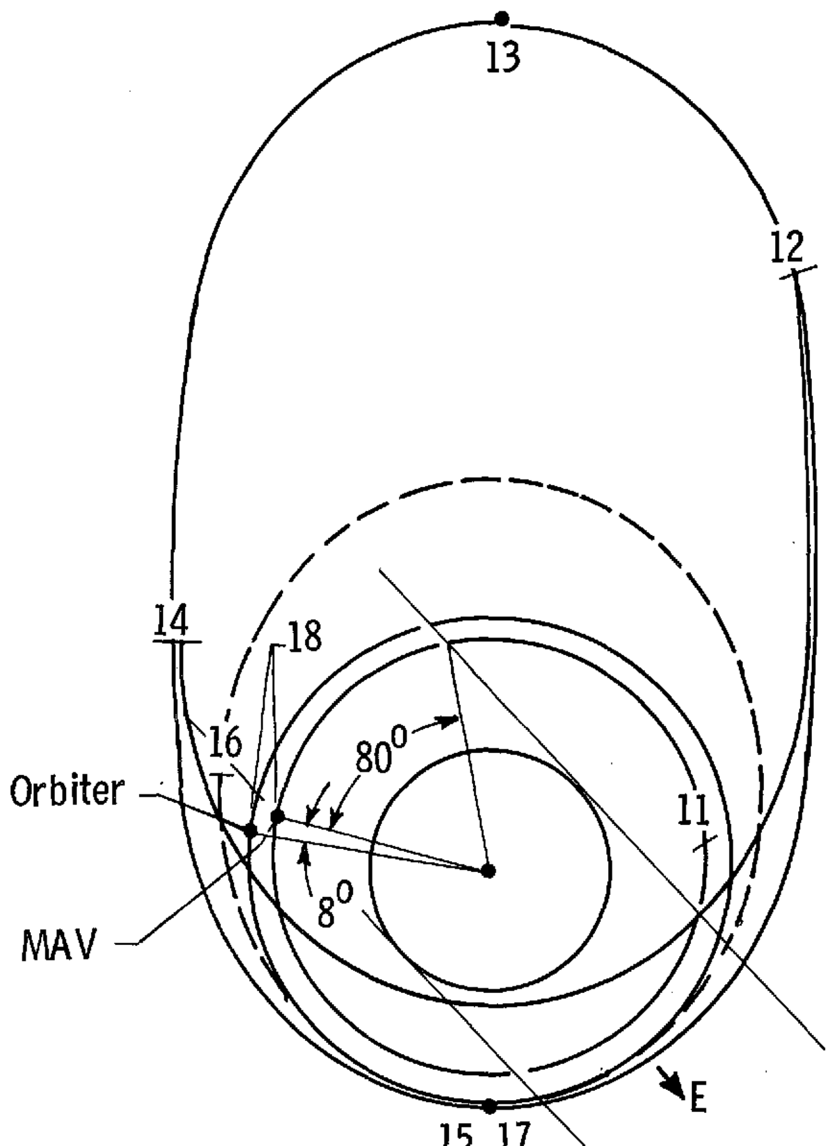
16) State vector update based on ~4 orbits of conventional doppler

17) Orbiter circularization (ΔV - 28.9 m/s) so that

18) 8^{o} phasing is achieved 1^{st} time out-of-shadow.

This phasing repeats in 19 MAV revolutions, since $P_o = (19/18)\, P_M$

MISSION SCHEMATIC #3 - ORBITER PERIAPSIS CHANGE
TO ORBITER CIRCULARIZATION TO 1st OCCULTATION EXIT



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MISSION SCHEMATIC #4 - FIRST OCCULTATION EXIT TO ORBITER DESCENT INITIATION

19) Simultaneous solution for orbiter and MAV states using conventional Doppler on orbiter and multi-vehicular ΔVLBI data (4 orbits of data; solution available at 8th orbit as shown).

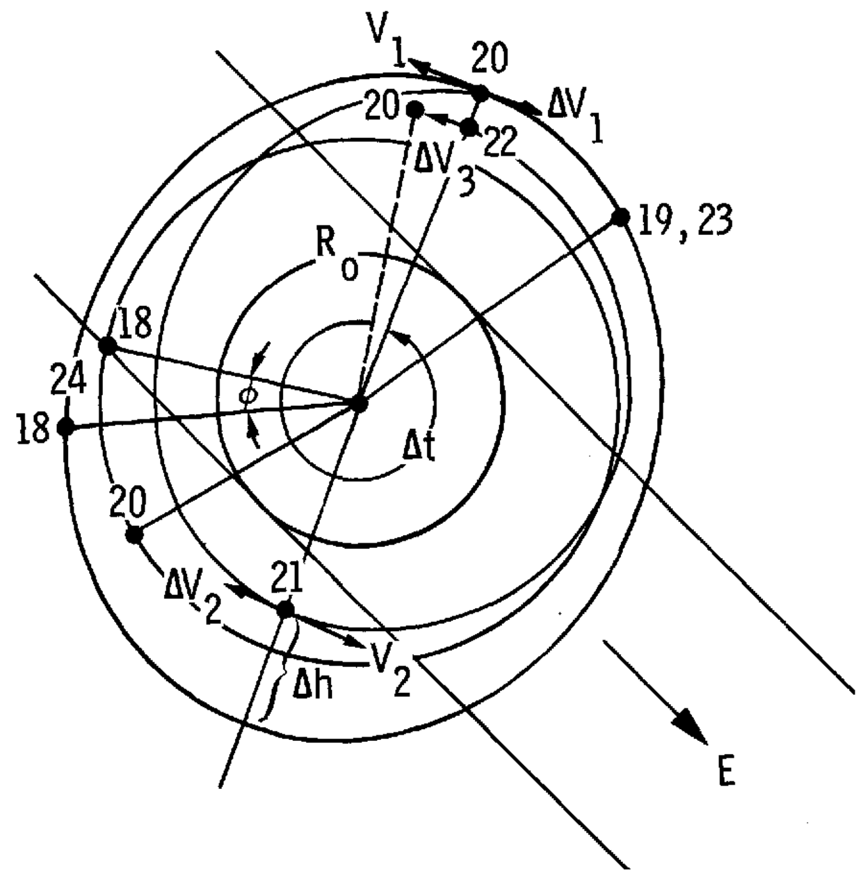
20) Propagate orbiter and MAV to time of next orbiter periapsis passage. Compute desired orbiter state (indicated by dotted line). Perform ΔV_1 to adjust apoapsis by Δh.

21) At a time Δt_1 later perform ΔV_2 to correct radius to R_o, 180° later.

22) At a time Δt_2 later perform ΔV_3 to recircularize at R_o the desired radius. Note that Δh was computed so that $\Delta t_1 + \Delta t_2 = \Delta t$.

23) Simultaneous solution for orbiter and MAV states based on orbiter conventional Doppler and multi-vehicular ΔVLBI data (based on 4 orbits data; solution available 4 orbits later at pt. 23).

MISSION SCHEMATIC #4 - FIRST OCCULTATION EXIT TO ORBITER DESCENT INITIATION



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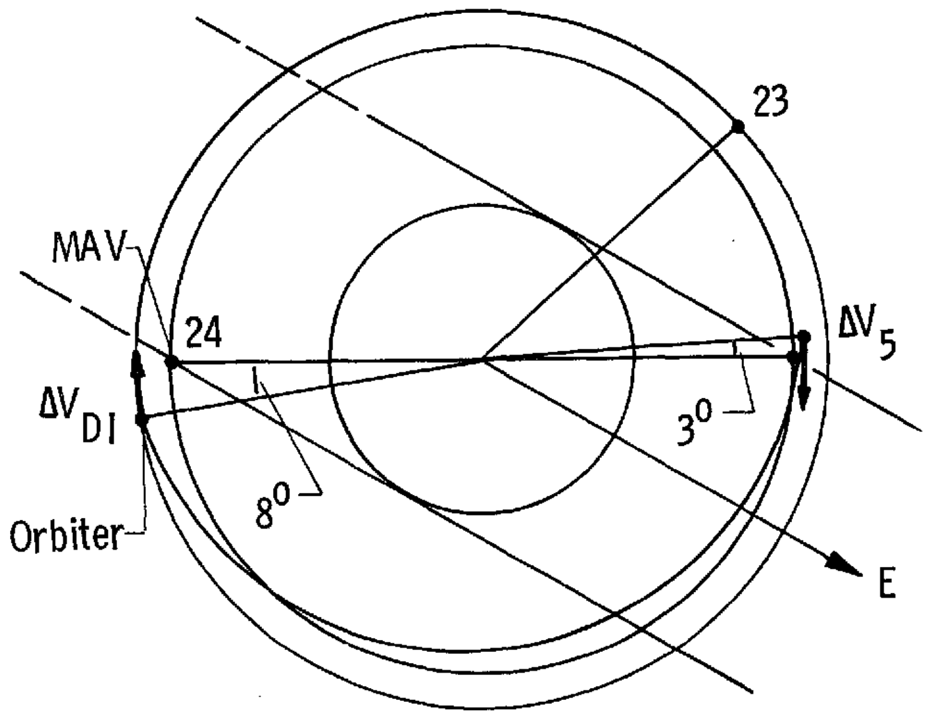
MISSION SCHEMATIC #5 - ORBITER DESCENT INITIATION (DI) TO TERMINAL RENDEZVOUS INITIATION (TRI)

24) Orbiter performs descent initiation maneuver ($\Delta V_{DI} \cong 24.2$ m/s) after exiting shadow for 18th time. This maneuver is a Lambert transfer to a target position 3° ahead of the MAV and ~50 km further out. The MAV meanwhile has traversed 180° of orbit so that DT for the Lambert is ½ a MAV orbital period = 1.76 hrs. Planar error is also taken out with this maneuver.

25) Orbiter performs terminal rendezvous initiation maneuver ($\Delta V_{TRI} \cong 24.5$ m/s) DT seconds later. At this time rendezvous radar acquisition of the MAV is made.

MISSION SCHEMATIC #5 -
ORBITER DESCENT INITIATION (DI) TO TERMINAL RENDEZVOUS INITIATION (TRI)



MISSION ΔV_{STAT} COMPONENTS

The total ΔV_{STAT} for the mission is the sum of the five ΔV_{STAT} contributors below. The total rendezvous ΔV_{STAT} is approximated as the sum of ΔV_{STAT}s for items 3), 4) and 5). The orbiter ΔV_{STAT} budget is the sum of ΔV_{STAT}s for items 1), 3) and 5). The MAV ΔV_{STAT} budget is that required by item 4).

MISSION ΔV_{STAT} COMPONENTS

1) Trans-Mars + Trans-Earth Midcourse Correction
2) Lander Deflection
3) Orbiter MOI and Circularization
4) MAV Ascent to MAV Circularization
5) Perfect Orbiter Insertion to Terminal Rendezvous Initiation

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ASSUMPTIONS FOR ΔV_{STAT} COMPUTATION FOR ORBITER MOI AND CIRCULARIZATION

This computation was performed assuming only encounter control and knowledge uncertainties and maneuver execution error. B-plane control and knowledge statistics are shown below. An optimal set of MOI burn controls (α, δ, t_B, TA) is computed based on the pre-encounter state estimate for each dispersed Monte Carlo case. The actual state is then integrated through the burn to produce the capture orbit. A similar technique is used to target the circularization burn except for this computation no knowledge error is assumed (i.e. estimate = actual state). Two post-circularization trims are computed to take out dispersions due to execution error. Statistics of total ΔV are computed for the three maneuvers and ΔV_{STAT} output as the 99 percentile sample.

ASSUMPTIONS FOR ΔV_{STAT} COMPUTATION FOR ORBITER MOI AND CIRCULARIZATION

Definition

$$\Delta V_{stat} = \text{99 percentile total } \Delta V - \text{Nom. } \Delta V$$

1) Representative Control & Knowledge Uncertainties Expressed in B-plane System

o Control: $\delta X_A = X_A - X_R$

o Knowledge: $\Delta X_E = X_E - X_A$

	$\sigma_{B \cdot R}$	$\sigma_{B \cdot T}$	θ_{SMAA}
Knowledge	210. km	60. km	90^0
Control	227. km	101. km	97^0

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ASSUMPTIONS FOR ΔV_{STAT} COMPUTATION FOR ORBITER MOI AND CIRCULARIZATION (concl)

2) Viking Execution Errors for MOI, CIRC

3) Finite Burn VITAP Optimization for MOI, CIRC Controls (α, δ, t_B, TA)
Target MOI to 1/a, CIRC to r_p.

4) Perfect In-Orbit O.D.

5) Trim to Desired Circular Radius (h_p = 2200)

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ΔV_{STAT} RESULTS FOR ORBITER MOI AND CIRCULARIZATION

The total ΔV_{STAT} for this mission segment is 53.3 m/s. The additional ΔV cost is incurred primarily from approach h_p dispersions and the effect of execution errors on the circularization maneuver.

ΔV_{STAT} RESULTS FOR ORBITER MOI AND CIRCULARIZATION

	99% ΔV	ΔV_{STAT}
*MOI	1144.5 m/s	28.7 m/s
HP	71.6 m/s	10.2 m/s
**CIRC	853.7 m/s	21.3 m/s
TRIM#1	18.0 m/s	18.0 m/s
TRIM#2	18.4 m/s	18.4 m/s
TOTAL ΔV	2066.1 m/s	53.3 m/s

* Due to h_p dispersion.

** Due to execution error.

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TRACKING SYSTEMS & DATA TYPES: ADVANTAGES & DISADVANTAGES

1. Conventional Doppler Data Types (DSN Single Vehicle Range, Range-Rate)

   Advantages: Utilize Existing System

   Disadvantages: Does Not Measure Inter-vehicular Quantities (e.g. relative range and/or range-rate)
   Slower Convergence of Relative State Error

2. Onboard Rendezvous Radar Range, Range-Rate

   Advantages: Required for Terminal Rendezvous Anyway
   Provides Direct Relative Data
   Rapid Solution for Relative State

   Disadvantages: Requires Proper Inter-vehicular Phasing (finite range)

continued

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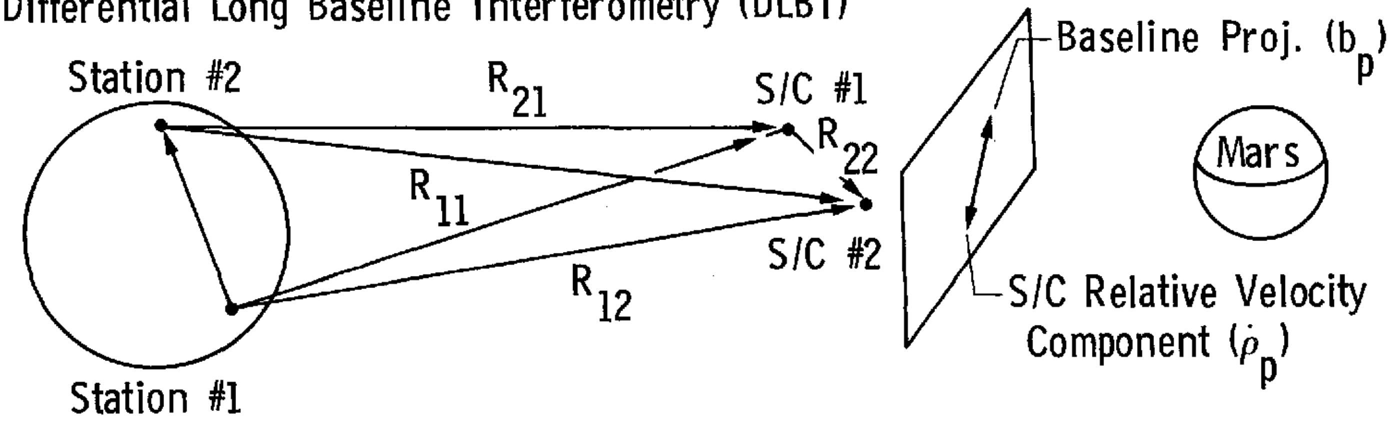
3. Differential Long Baseline Interferometry (DLBI)



Advantages: Flexible Inter-vehicle Phasing Requirement
Rapid Relative State Solution (measures relative velocity component "directly")

Disadvantages: Implementation Cost
1) Simultaneous Data Differencing
2) DPODP Modification

ΔVLBI VS CONVENTIONAL DOPPLER

Conventional Doppler will yield a relatively accurate intervehicular state after many orbits of data. This time is required for correlations to build up. ΔVLBI affords a much quicker, more accurate solution because it measures a component of the relative velocity directly.

ΔVLBI VS CONVENTIONAL DOPPLER

Relative State Accuracy: Single Vehicle Doppler Tracking

Time	RSS S/C #1	RSS S/C #2	RSS Rel.
1 rev	168.3 km/130.7 m/s	88.6 km/57.7 m/s	193.0 km/139.4 m/s
2 revs	90.0 km/ 37.8 m/s	41.1 km/48.7 m/s	68.7 km/ 19.5 m/s
3 revs	63.0 km/ 14.9 m/s	37.0 km/ 6.1 m/s	28.7 km/ 4.3 m/s
4 revs	9.9 km/ 1.3 m/s	5.8 km/ .8 m/s	4.9 km/ .7 m/s

Relative State Accuracy: DLBI Tracking

Time	σ_x (km)	σ_y (km)	σ_z (km)	$\sigma_{\dot{x}}$ (m/s)	$\sigma_{\dot{y}}$ (m/s)	$\sigma_{\dot{z}}$ (m/s)	RSS
1/8 rev	24.7	29.3	15.8	6.3	9.1	2.7	41.5/11.4
1/4 rev	1.1	2.4	1.1	.6	1.0	.8	2.9/ 1.4
3/8 rev	.9	1.0	.8	.6	.7	.4	1.6/ 1.0
1/2 rev	.7	1.1	.7	.2	.5	.3	1.5/ .6
1 rev	.5	.5	.2	.1	.2	.1	.7/ .3

PREDICTION CAPABILITY FOR h_p = 2200 CIRCULAR

This figure shows the O.D. capability with Mariner 9 derived Mars gravity harmonics and associated uncertainties. Four orbits of data is sufficient at this circular altitude.

PREDICTION CAPABILITY FOR h_p = 2200 CIRCULAR

REVS TRACKING	REVS PREDICTION	PER (SEC)	HP (KM)	INC (DEG)	NODE (DEG)	OMEGA (DEG)	T≠TP (SEC)	RSS POS (KM)	RSS VEL (M/S)
4	0	.254	.0311	.0264	.0032	1.804	63.83	.912	1.35
4	1	.233	.0206	.0264	.0047	2.028	72.10	1.248	1.406
4	2	.137	.0187	.0268	.0062	1.429	50.82	1.473	1.470
4	3	.197	.0295	.0267	.00776	1.932	68.62	1.633	1.506
4	4	.576	.0580	.0264	.00928	6.998	247.77	1.949	1.573
4	5	.969	.119	.0266	.0108	12.44	440.12	2.259	1.67
4	6	1.434	.2018	.0268	.0124	19.32	683.24	2.357	1.71
4	7	2.110	.337	.0264	.0140	32.66	1154.13	2.617	1.78
6.6	0	.2606	.0416	.0069	.0052	.679	23.79	.605	.265

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ASCENT, RENDEZVOUS AND DOCKING GUIDANCE AND CONTROL

Rendezvous Radar - W. Koppl

F. A. Vandenberg

PRELIMINARY G&C BASELINE DEFINITION

The salient features of the MAV flight are shown in this Vugraph. During the landed phase, the lander azimuth and latitude are determined by gyrocompassing utilizing the lander inertial sensors. The sun sensors and the lander-to-MAV encoders are used to determine lander attitudes and longitude, and MAV attitudes. The MAV is erected and launched in the preferred injection orbit and at an optimum launch attitude. The MAV is three-axis stabilized and uses open loop guidance with a constant pitch-over rate during ascent phase. The ignition of the second stage, that accomplishes the initial orbit injection, is executed on time based on the MAV clock. The MAV attitudes during orbital operations are determined by the sun sensors and Earth direction as determined by the MAV Earth pointing system. In orbit, the MAV and orbiter state is determined by DSN tracking. The MAV utilizes a proportional navigation type of rendezvous guidance, that has been simplified, to accomplish the terminal rendezvous. A combination rendezvous and docking CW system is suggested that is modulated by tones.

PRELIMINARY G&C BASELINE DEFINITION

Landed Phase

Gyrocompassing Used to Determine Lander Latitude and Azimuth
Sun Sensor and Lander-to-MAV Encoder Used to Determine
  Lander and MAV Attitudes
  Lander Longitude
MAV Erected and Launched in the Preferred Injection Orbit and
  at the Optimum Attitude

Ascent Phase

Three Axis Stabilized
Open Loop Guidance with Constant Pitch Rate
Initial Orbit Injection Based on MAV Clock

Initial Rendezvous Phase

MAV In-Orbit Attitude Determined by Sun Sensor and Earth Direction
MAV and Orbiter State Determined by DSN Tracking

Terminal Rendezvous Phase

Modified Proportional Navigation Terminal Rendezvous Guidance
Combination Microwave Rendezvous and Docking Radar

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LANDED POSITION AND ATTITUDE UPDATE

The uncertainty of the position of the lander can be determined in three ways; namely, gyrocompassing, landing footprint accuracy, and Earth-based tracking. Gyrocompassing to determine the vehicles position is accurate enough to accomplish the rendezvous. Since we have an S-band system aboard, Earth-based tracking will be used to determine the vehicles position very accurately and will reduce the errors that the terminal rendezvous system has to take out. During the first half of the study and in the table that describes the launch phase errors, gyrocompassing was used, which takes about 5 minutes to perform.

LANDED POSITION AND ATTITUDE UPDATE

Position

Gyrocompassing

Latitude - 5^0 (3σ) } Lander IRU
Azimuth - 5^0 (3σ) }

Earth Based Tracking

Latitude - 0.3546^0 (21 km - 3σ)
Longitude - 0.03039^0 (1.8 km - 3σ)
Altitude - 984.24 ft (0.3 km - 3σ)

Landing Accuracy (650 x 1748 km)

Latitude - 10.97^0 (650 km - 3σ)
Longitude - 29.5^0 (1748 km - 3σ)

Attitude

Launch Ramp Angle - Sun Sensor

Launch Attitude - Sun Sensor

Lander Attitude - Sun Sensor & MAV-to-Lander Encoder

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LAUNCH PHASE ERROR SOURCE

The launch phase errors were derived based on the error sources described in this Vugraph, and the landed position and attitude errors. The amplitude of this error is defined by the right hand column of this figure. The launch phase errors used in this study are shown in one of the later Vugraphs.

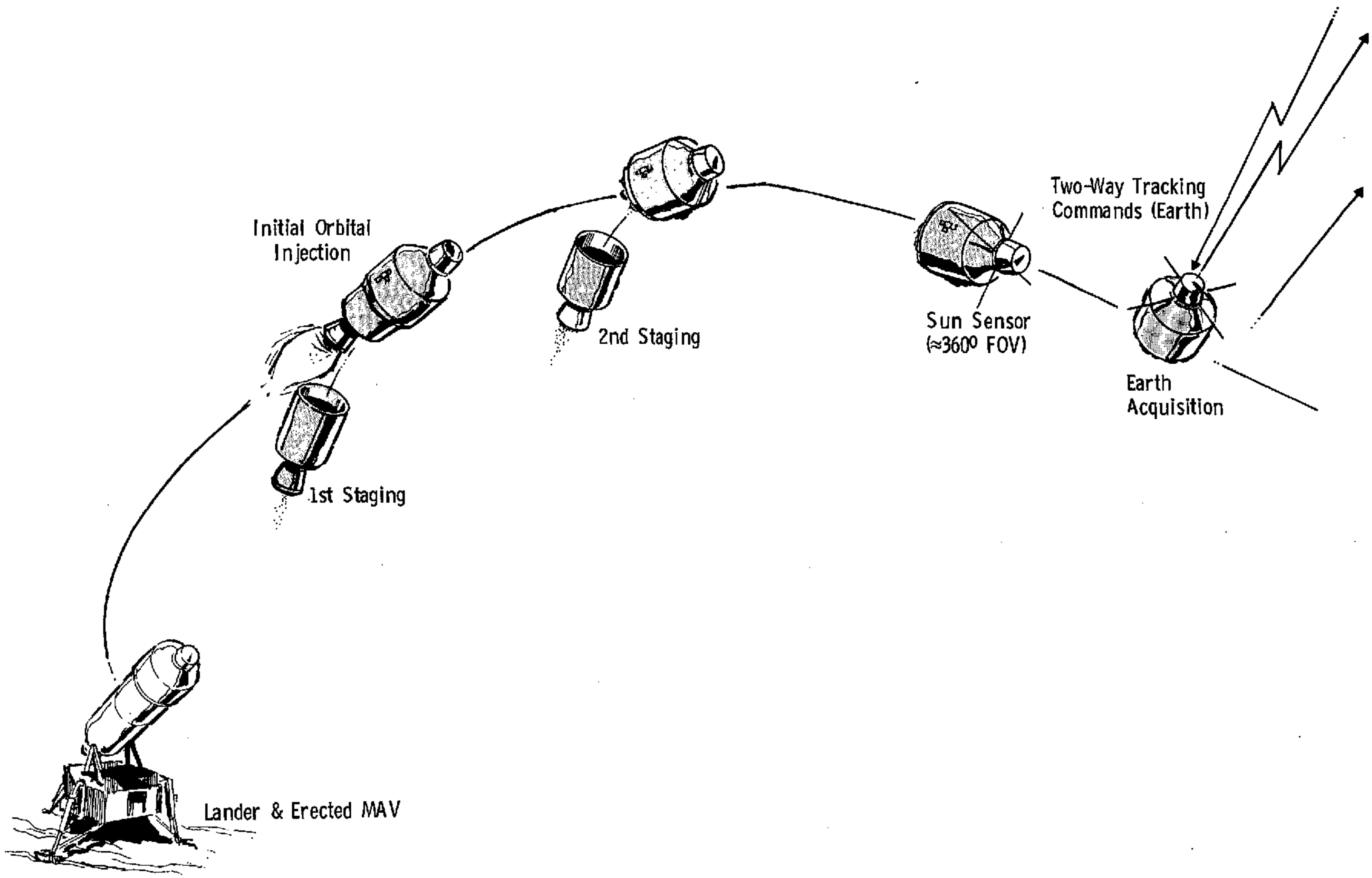
LAUNCH PHASE ERROR SOURCE

Error Source	Error
Pitch Rate	Gyro Bias Error
Liftoff Weight	0.3% (3)
Ramp Angle	MAV Sun Sensor (0.25^{0})
J2 Gravity Coefficient	Viking '75 Environmental Specs
Central Gravity Coefficient	Viking '75 Environmental Specs
Launch Azimuth	MAV Sun Sensor (0.25^{0})
Coast Time	Launch Thrust Ignition Errors
Launch Site Altitude	Based on Estimated Lander Position
Geodetic Latitude	5.0 deg (3σ)
Propellant Weight	0.25 (3σ)
Burn Time	4.0% (3σ)
Thrust	4.0% (3σ)
Impulse	0.75% (3σ)

LAUNCH, ASCENT AND EARTH ACQUISITION

Before launch, the MAV is erected to the azimuth of the ejection orbit and to the optimum initial pitch attitude for the gravity turn it is supposed to execute. The MAV is three-axis stabilized and uses open loop guidance during ascent with a constant pitch-over rate. The dynamic pressure and pitch profile used during ascent is shown in a backup vugraph. The second stage, which injects the MAV into a 100 km x 2200 km orbit, is ignited by a time discrete from the control computer. Shortly after orbital injection, the MAV is commanded by stored command to point toward Earth and turn-on the MAV Earth pointing system. The MAV has the option on command of controlling its attitudes with a Sun sensor system and Earth sensing system or the Sun sensor system and a pitch rate gyro.

LAUNCH, ASCENT, AND EARTH ACQUISITION



THREE AXIS STABILIZED MAV

The suggested G & C system for the MAV is shown with the estimated weight and power requirement. This vugraph has all the components needed to compare the spin-stabilized system to the three-axis stabilized system. The components, which includes the required maneuver propellant, of the three-axis stabilized MAV weighs about 2 kgms more than the spin stabilized vehicle. The propellant needed to counteract the thrust offset adds to total ΔV, so this propellant is not lost. The weight and power requirements for the spin-stabilized vehicle is described in one of the backup vugraphs. The three-axis stabilized system was selected for this mission as it does the job better and has comparable weight.

THREE-AXIS STABILIZED MAV

Components	Weight	Power
4 ΔV & Launch ACS Engines	2.18 kg (4.8 lbs)	------
8 ACS Engines	1.45 kg (3.2 lbs)	------
1 All Attitude Sun Sensor System	0.18 kg (0.4 lbs)	------
3 Rate Gyros & Electronics	*1.36 kg (3.0 lbs)	5.0 watts
1 Axial Accelerometer	*0.14 kg (0.3 lbs)	------
1 Computer & Sequencer	*1.59 kg (3.5 lbs)	4.0 watts
1 Transponder & Antenna Feed	*1.59 kg (3.5 lbs)	20.5 watts
1 Antenna Dish & Reflector	0.52 kg (1.2 lbs)	------
	9.01 kg (19.9 lbs)	

ACS Propellant Required - 4.2 lbs (Isp = 235 sec)

44 N (10 lb) Ascent Engines for Thrust Offset and ΔV

* Uncased

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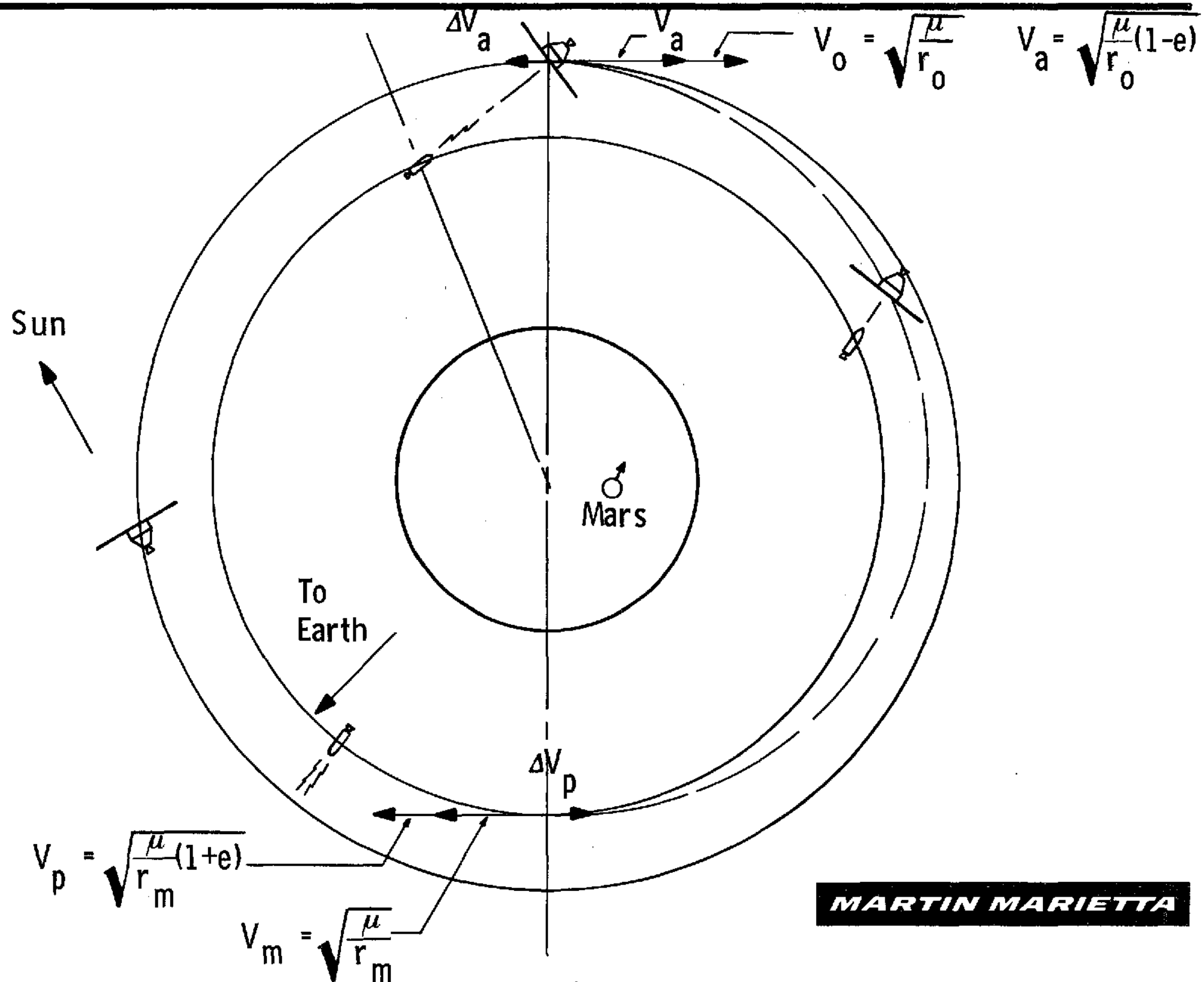
TERMINAL RENDEZVOUS EXECUTION

The orbital geometry for initial and terminal rendezvous is shown. During the initial rendezvous phase and when the vehicles are not executing a maneuver sequence, the MAV is pointed toward the Earth and the orbiter is Sun-Canopus oriented. When the rendezvous radar is within acquisition range, the MAV and the orbiter are commanded to point at each other by stored commands in the control computers and executed like any other orbital meanuvers. During terminal rendezvous, the orbiter attitudes are controlled by the rendezvous system and the MAV attitudes are controlled by the MAV pointing systems, which point both vehicles along the vehicles line-of-sight (LOS). Terminal rendezvous should be executed in approximately one half of an orbital period to be efficient, i.e., to approximate a Hohmann transfer. The terminal rendezvous will always be compared to a Hohmann transfer (two impulse transfers) to check its efficiency.

TERMINAL RENDEZVOUS EXECUTION



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TYPES OF RENDEZVOUS AND INTERCEPT GUIDANCE SCHEMES

This Vugraph shows some of the common type of guidance schemes that are considered for intercept and rendezvous. A constant bearing course type of guidance is generally considered the best algorithm for a rendezvous vehicle with a limited amount of thrust. This type of algorithm is the only one that has been implemented in spacecrafts in our space program. Proportional navigation guidance in a practical algorithm to implement a constant bearing course.

TYPES OF RENDEZVOUS AND INTERCEPT GUIDANCE SCHEMES

Pursuit Course

Modified Pursuit Course

Janus Beam Rider

Constant Bearing Course

Proportional Navigation

Modified Proportional Navigation

Optimum Guidance Scheme

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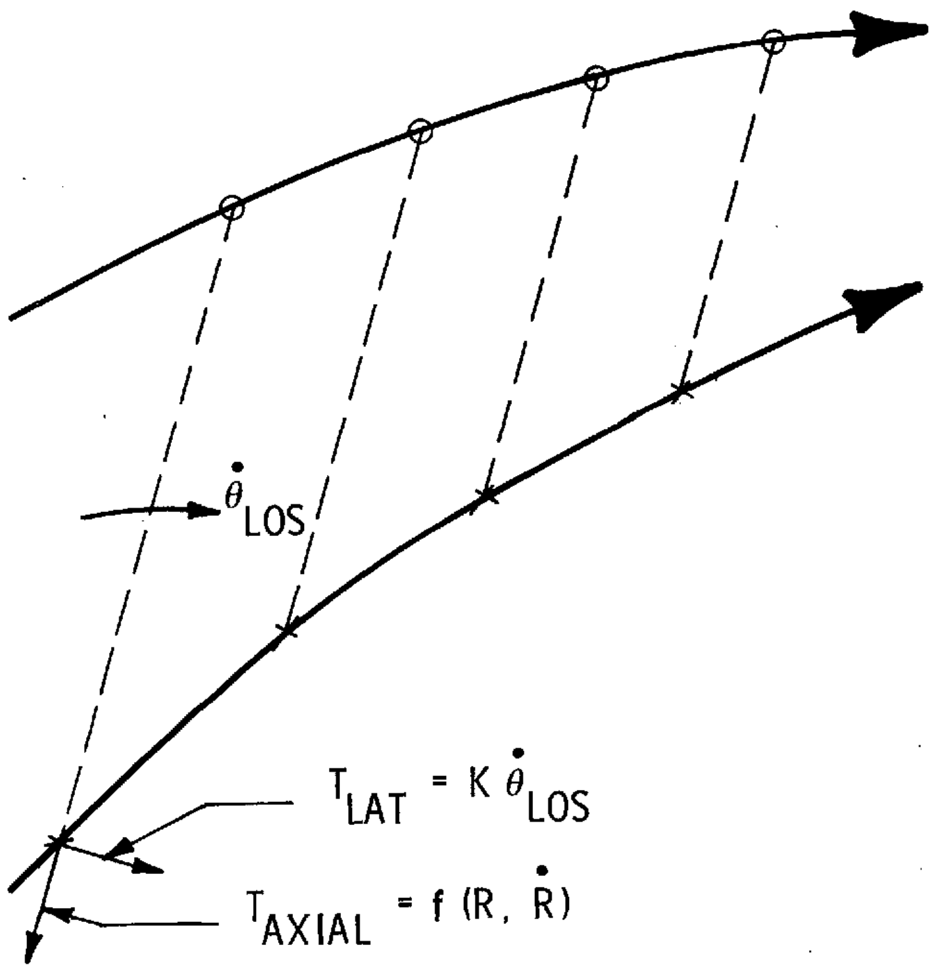
PROPORTIONAL NAVIGATION

The left hand figure shows how an intercept--where vehicles positions are matched--is accomplished. If the line-of-sight rate is kept small--LOS angles are constant--an interception will be accomplished. The relative positions and velocities have to be driven to zero simultaneously to accomplish a rendezvous between the two vehicles. The proportional navigation guidance is implemented by axial and lateral control equations. The lateral thrusters keep the LOS rates small. The axial thrust algorithm commands the vehicle range and range rate to zero simultaneously. A simplified version of proportional navigation guidance has been baselined in this study, which we call a modified proportional navigation guidance. The axial control law is designed, so the LOS rate is kept small throughout the terminal rendezvous.

PROPORTIONAL NAVIGATION



Proportional Navigation

$T_{AXIAL} = f\,(R,\ \dot{R})$

$T_{LAT} = K\,\dot{\theta}_{LOS}$

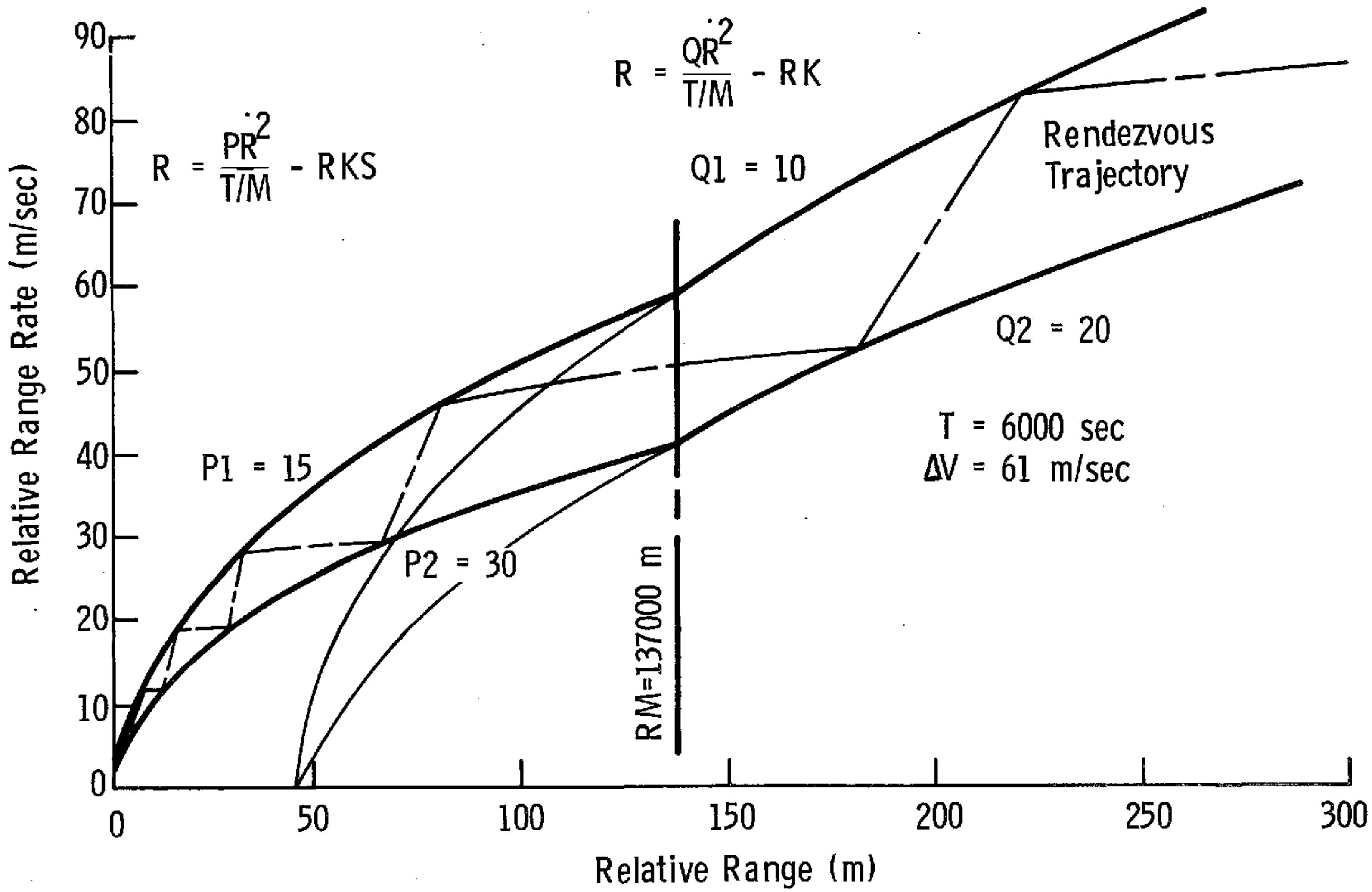
Modified Proportional Navigation

$T_{AXIAL} = f\,(R,\ \dot{R})$

AXIAL THRUST CONTROL CURVES

The axial thrust control curves, that are used, are shown here. Two sets of control curves are used, one is used above the gain change altitude R_M and another is used below this altitude. The control gains, Q, are used above the gain change altitude and control gains, P, are used below that altitude. The switching lines P_1 and Q_1 turn the thrust on and the switching lines P_2 and Q_2 turn the thrust off. The vehicle switches to a docking algorithm when the relative range is less than 30 m. This algorithm commands the vehicle to close at a constant velocity, while the vehicles are pointing down the line-of-sight.

AXIAL CONTROL CURVES (MPN)



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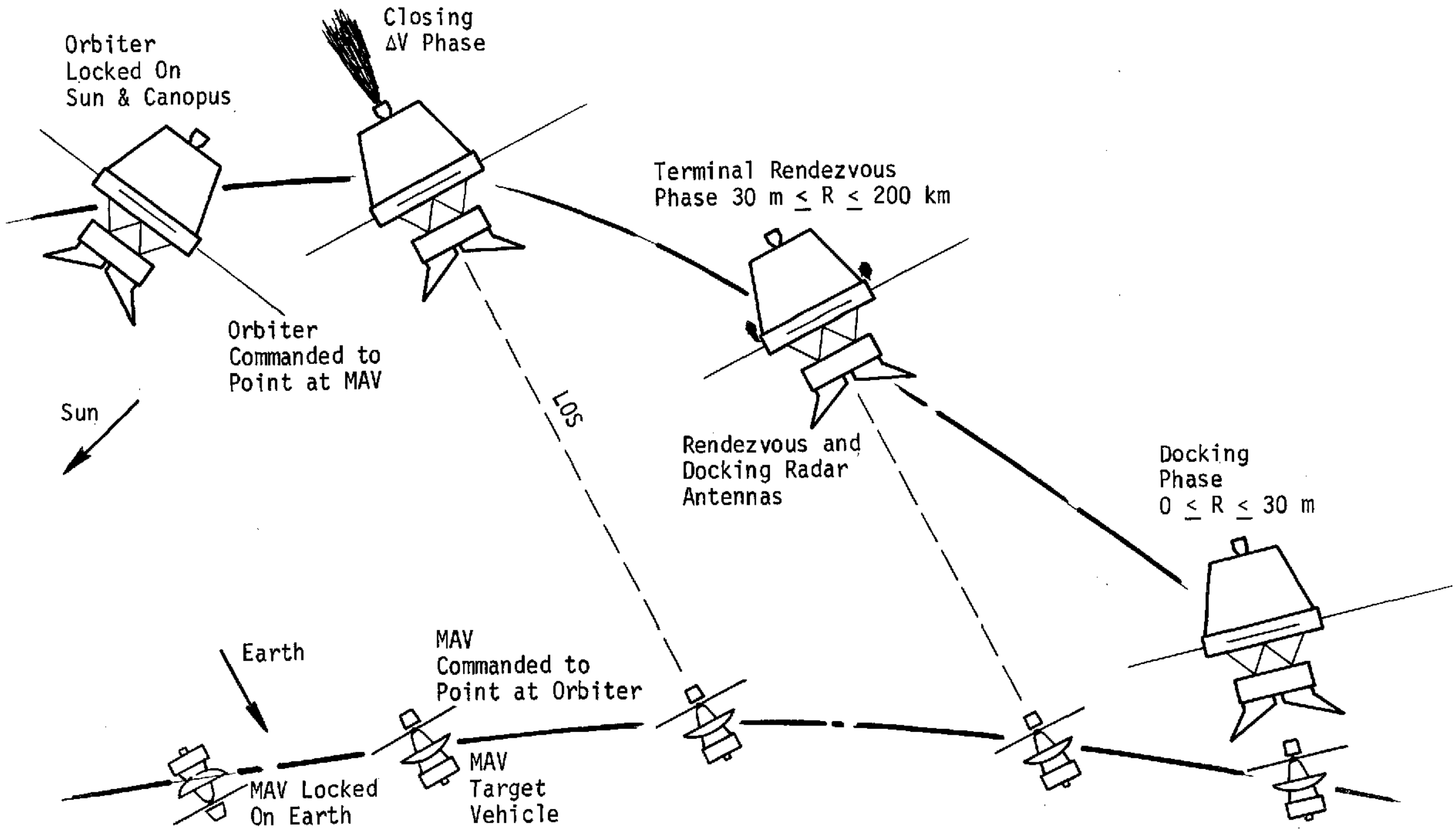
RENDEZVOUS AND DOCKING PHASES

Just before the vehicles are within the rendezvous radar acquisition range, the vehicles are commanded to point at each other. The orbiter executes a predetermined closing ΔV maneuver that imparts a closing velocity to the orbiter. The orbiter and MAV attitudes are always controlled to point along the LOS. The vehicle axial thrust is controlled by the axial control law. The docking algorithm is used when the vehicles range is less than 30 m.

RENDEZVOUS AND DOCKING PHASES



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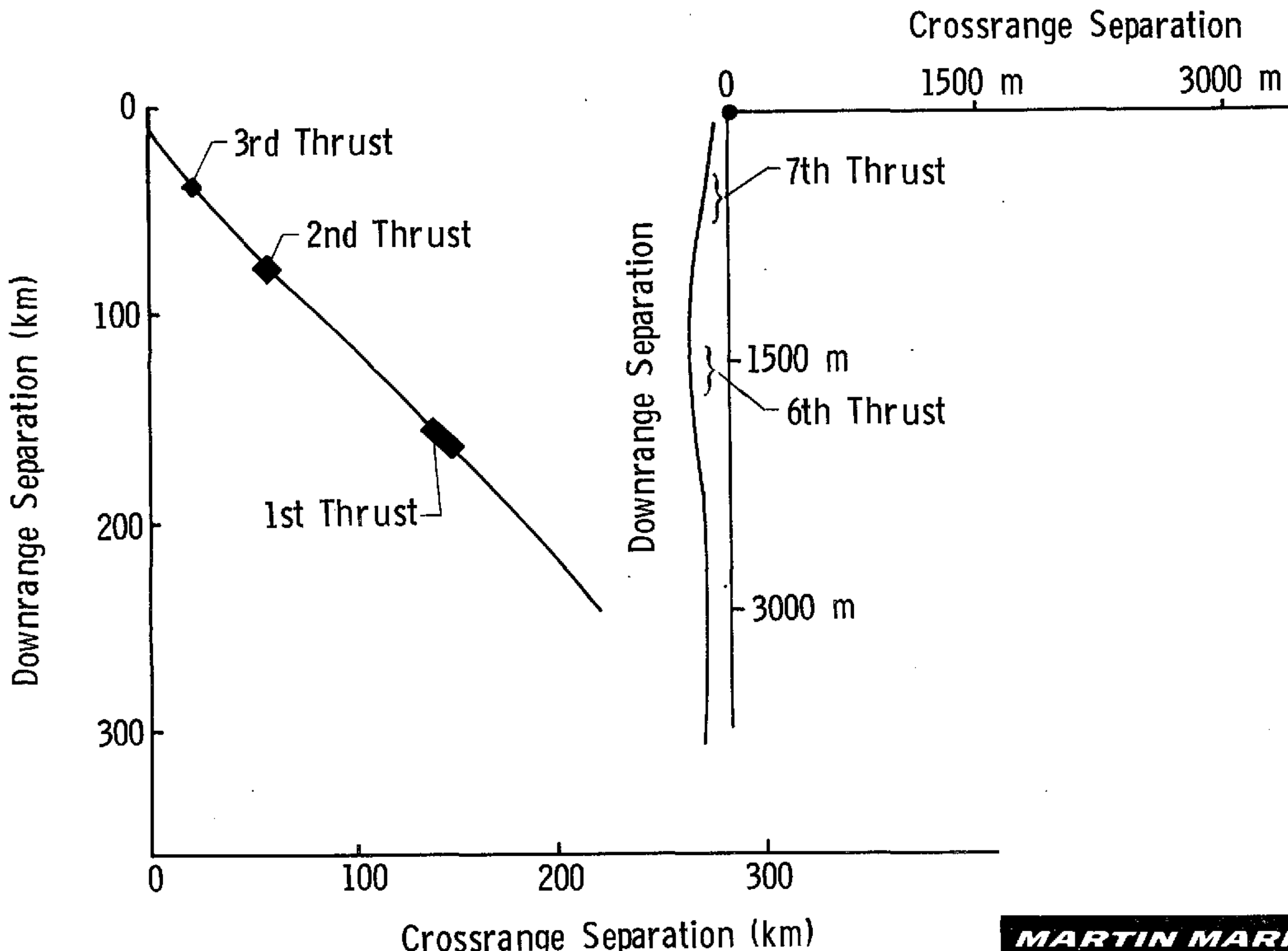
RENDEZVOUS TRAJECTORY

This Vugraph shows the rendezvous trajectory in the MAV centered coordinate system. This figure shows the thrust periods and the vehicle rendezvous trajectory to the target vehicle (MAV). The vehicle's rendezvous at close range is shown in the insert on the left of the figure.

RENDEZVOUS TRAJECTORY



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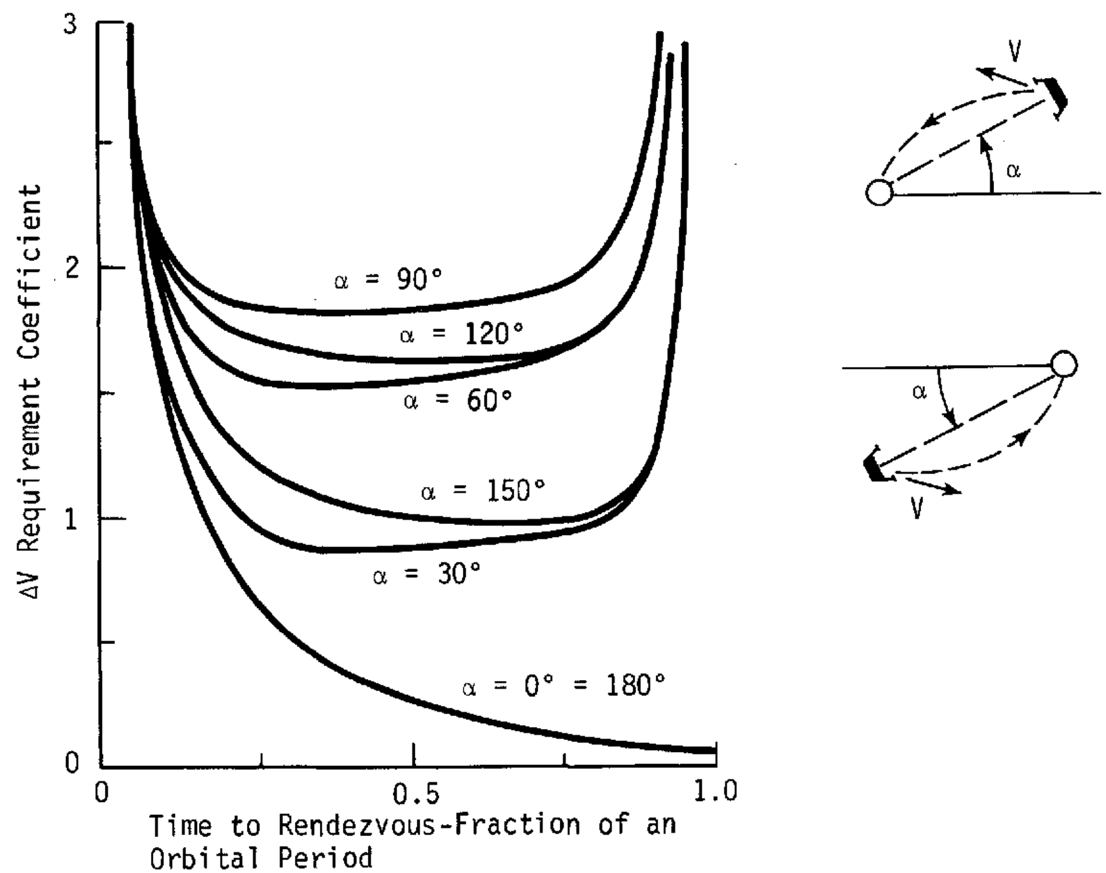
RENDEZVOUS PROPELLANT EFFICIENCY

This Vugraph shows how the time of rendezvous affects the propellant efficiency. The ΔV requirement coefficient shown as the ordinate of the figure is proportional to the amount of propellant needed above the most optimum case. When $\alpha = 0$ or 180 degrees, where the vehicle is essentially in the same orbit, the most efficient rendezvous can be achieved. When $\alpha = 90$ degrees -- the vehicle is in a larger or smaller orbit -- the vehicle uses the most propellant. The reason these rendezvous are so inefficient is that a large closing ΔV is needed to catch the satellite which has to be taken out during the terminal rendezvous phase.

RENDEZVOUS PROPELLANT EFFICIENCY



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TERMINAL RENDEZVOUS MALFUNCTION OPTIONS

With no malfunction, a cooperative rendezvous can be executed from a maximum range of 1000 km with 4.0 watts of output power from the MAV transponder and 25.0 watts average output power from the rendezvous radar. If the orbiter propulsion system fails, the MAV can rendezvous with the orbiter by executing axial thrust commands calculated on the orbiter and fed to the MAV over the command link. If the transponder transmitter fails a passive cooperative rendezvous can be achieved from 36.0 km. The passive cooperative rendezvous uses the MAV antenna to reflect the microwave signal passively back to the orbiter. If the transponder receiver fails or the whole transponder fails, the vehicle can be skin tracked -- non-cooperative rendezvous -- if the MAV is closer than 5 km.

TERMINAL RENDEZVOUS MALFUNCTION OPTIONS

NO MALFUNCTION (COOPERATIVE RENDEZVOUS)

ORBITER RENDEZVOUS WITH MAV

$R_{MAX} = 1000$ km

$P_{MAV} = 4.0$ watts (Average Power)

$P_O = 25.0$ watts (Average Power)

ORBITER PROPULSION SYSTEM FAILURE (COOPERATIVE RENDEZVOUS)

MAV RENDEZVOUS WITH ORBITER

MAV AXIAL THRUSTER COMMANDED OVER MAV-ORBITER COMMAND LINK

$R_{MAX} = 1000$ km

MAV TRANSPONDER TRANSMITTER FAILURE (PASSIVE COOPERATIVE RENDEZVOUS)

ORBITER RENDEZVOUS WITH MAV

OR

MAV RENDEZVOUS WITH ORBITER

$R_{MAX} = 35.6$ km

TRANSPONDER RECEIVER FAILURE (NON-COOPERATIVE RENDEZVOUS)

ORBITER RENDEZVOUS WITH MAV

$R_{MAX} = 5$ km

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LAUNCH PHASE ERRORS

This vugraph shows the launch phase errors as derived for this study. The errors were used as input parameters to the simulation program to determine ΔV stat and the terminal rendezvous initiation state errors.

LAUNCH PHASE ERRORS

Error Source	Nominal	Error (1σ)
Pitch Rate (1)	0.20612 deg/sec	0.00416 deg/sec
Pitch Rate (2)	0.20612 deg/sec	0.00416 deg/sec
Liftoff Weight	250 kg (550 lbs)	0.25 kg
Ramp Angle	39.388 deg	0.25 deg
J2 Gravity Coefficient	0.00197	0.67 (10^{-5})
Central Gravity Coefficient	42828.4 km^3/sec^2	0.467 km^3/sec^2
Launch Azimuth	----	0.25^0
Coast Time	198.6 sec	0.045 sec
Launch Site Altitude	----	608.3 m (1995.7 ft)
Geodetic Latitude	0 deg	5.0 deg
Weight Propellant (1)	95.9 kg (211.0 lbs)	0.266 kg
Burn Time (1)	41.5 sec	0.554 sec
Thrust (1)	5280 N (1200 lbs)	5.3 N
Weight Propellant (2)	87.7 kg (193.0 lbs)	0.243 kg
Burn Time (2)	75.8 sec	1.01 sec
Thrust (2)	2640 N (600 lbs)	6.6 N
Impulse (1)	219120 N-sec	1643.3 N-sec
Impulse (2)	200112 N-sec	1500.0 N-sec

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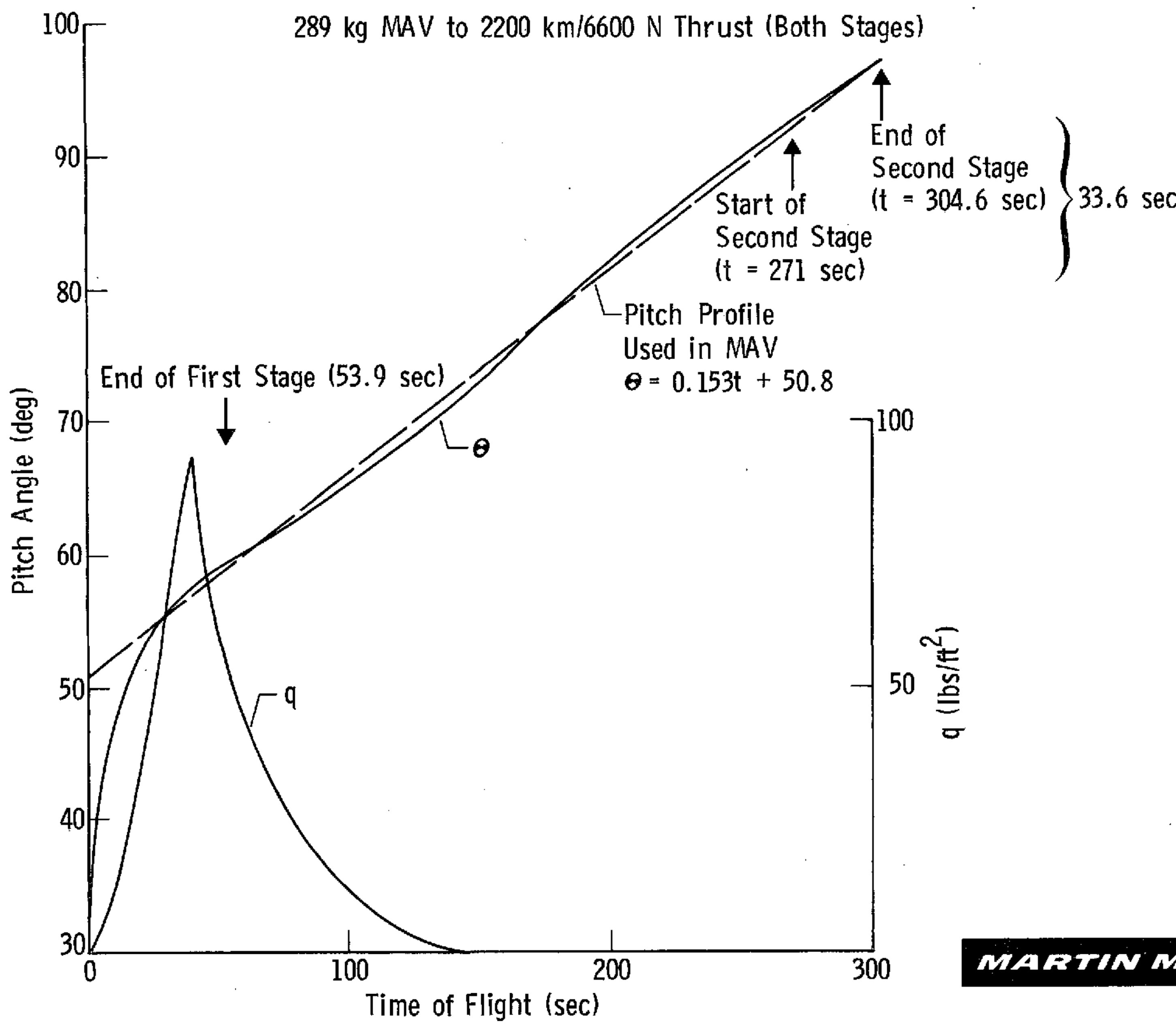
PITCH ANGLE AND DYNAMIC PRESSURE VS TIME OF FLIGHT

The dynamic pressure and pitch profile for a vehicle following a gravity turn is shown as a function of time of flight. A constant pitch-over rate of 0.15 degrees/second and initial pitch angle of 50.8 degrees are used in the MAV for this study. The MAV has a 54 second first stage, 217 second coast period and 34 second stage.

PITCH ANGLE AND DYNAMICS PRESSURE VS TIME OF FLIGHT



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ATTRIBUTES OF THREE-AXIS VS SPIN STABILIZED MAV

The attributes of the three-axis stabilized and spin stabilized systems are discussed. Generally speaking the three axis stabilized system excells where you have a relative short mission with many maneuvers that have to be executed very accurately. The spin stabilized spacecraft excells for long missions with few maneuvers that can be executed with open loop maneuvers.

ATTRIBUTES OF THREE-AXIS VS SPIN STABILIZED MAV

Three-Axis Stabilized	Spin Stabilized
Attitude maintained by slightly heavier subsystems that continually consume power.	Attitude maintained automatically at no expense of power on weight of auxiliary subsystems.
More efficient at attitude maneuver.	Less efficient at attitude maneuvers.
Optimum system for missions requiring many attitude reorientations.	Optimum system for long missions requiring few attitude reorientations.
Less sensitive to dynamic imbalance.	More sensitive to dynamic imbalance.
Higher power requirements.	Probably lower overall power requirements.
Does not provide sensor scanning.	Does provide sensor scanning.
Less complex computations to determine inertial attitude.	Complex calculations required for attitude determination.
Closed loop maneuvers.	Open loop maneuvers.
Requires more complex thermal protection.	Good thermal characteristics.
G&C hardware for rendezvous and docking is simpler.	Maneuvers must be executed in a rotating coordinate frame.

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ATTRIBUTES OF THREE-AXIS VS SPIN STABILIZED MAV (concluded)

Three-Axis Stabilized	Spin Stabilized
Easier to analyze during development.	More costly developmental analysis.
ACS system must correct for thrust.	Minimizes thrust offsets.

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SPIN STABILIZED MAV

The weight and the power comparison for the spin stabilized MAV is shown with all the components, that are needed for this comparison. The amount of propellant, that is needed to precess the spin stabilized vehicle 720 degrees for the required maneuvers, is shown.

Components	Weight	Power
4 Pitch & Yaw Engines	.73 kg (1.6 lbs)	----
2 Roll Maintenance Engines	.36 kg (0.8 lbs)	----
1 Sun Sensor	0.09 kg (0.2 lbs)	----
1 Axial Accelerometer	0.23 kg (0.5 lbs)	----
1 Computer & Sequencer	2.73 (6.0 lbs)	4.0 watts
1 Transponder	1.81 kg (4.0 lbs)	4.0 watts
1 Antenna	0.68 kg (1.5 lbs)	----
	6.6 kg (14.6 lbs)	

ACS Propellant Required - 6.0 lbs (I_{sp} = 235 sec)

Maneuvers:		
Launch	100^0	
Initial Rendezvous	480^0	
Terminal Rendezvous	100^0	
Contingency	60^0	(10%)
	720^0	

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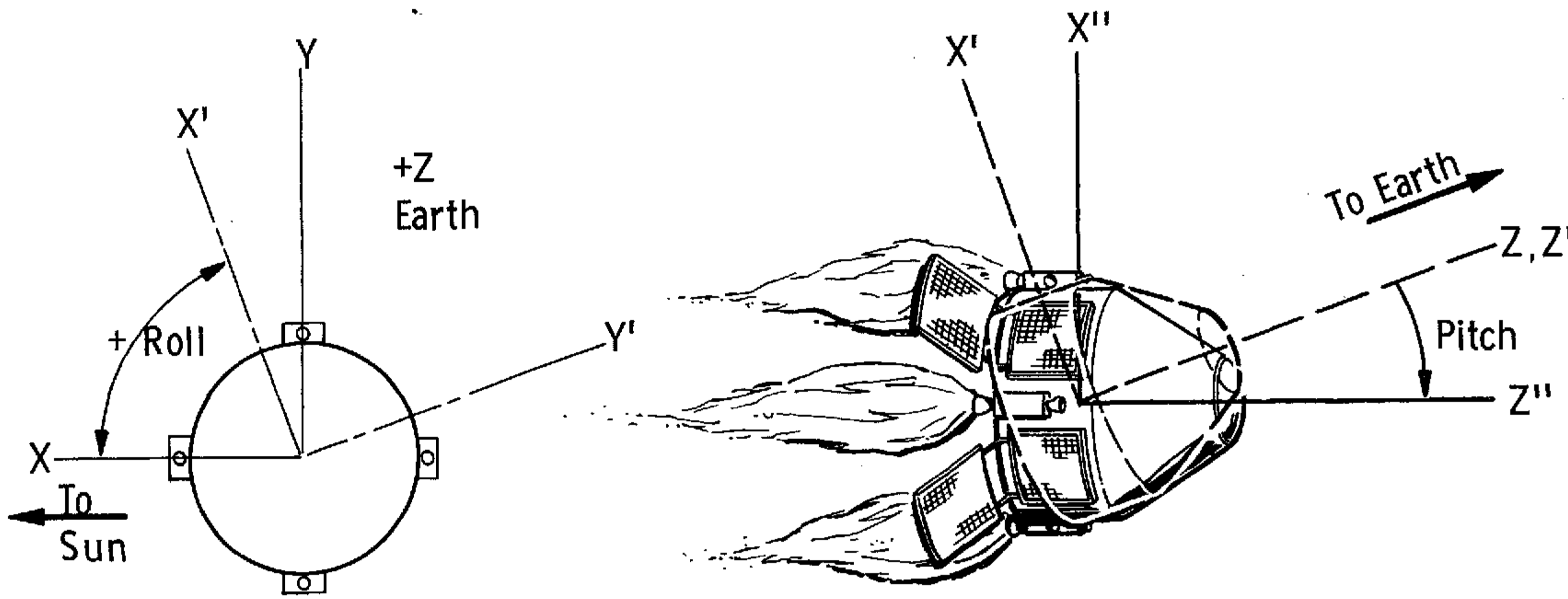
MAV ORBITAL MANEUVER

This vugraph shows how the MAV executes orbital maneuvers. The rate gyros are turned on prior to execution of the maneuver so they can be warmed up. Initially, the vehicle is commanded to roll about the MAV-Earth line until the desired ΔV direction is in the pitch plane, while the vehicle is still pointing at Earth and can receive commands. The executed roll maneuver can be verified by the Sun sensor system. The vehicle then pitches to the desired maneuver attitude and can again be verified by the Sun position as sensed by the sun sensors. After the maneuver is executed, the engines are shut down when the required ΔV is achieved as sensed by the axial accelerometer. The vehicle then returns to the Earth pointing orientation by executing the attitude maneuvers in the reverse order.


Y
X'
+Z
Earth
+ Roll
Y'
X
To
Sun
X'
X''
To Earth
Z,Z'
Pitch
Z''

LASER RADAR RANGE EQUATION

The equation describing the current signal-to-noise ratio in the photomultiplier tube of the receiver is shown. This equation can be used to determine the maximum range capability of a laser rendezvous system. The advantage of this type of system is that it can be used both for terminal rendezvous and docking. This type of system is also very accurate when compared to other types. The laser radar is generally heavier than the microwave system and is presently a laboratory curiosity and would be expensive to develop in a flight qualified article.

LASER RADAR RANGE EQUATION

$$\frac{i_s}{i_n} = \frac{P_t d_r d_c^{\,4} K^{1/2}}{R^4 \, \theta_r \, \theta_t^{\,2} \, \lambda^2 \left[1.125 \pi e \Delta F \Delta\lambda N_b\right]^{1/2}}$$

d_M = MAV corner reflector dia = 10 cm

d_r = orbiter receiver aperture = 8.75 cm

θ_t = orbiter aperture beamwidth = 0.03^0

θ_r = receiver FOV = 0.03^0

P_t = transmitter peak power

λ = wavelength = $0.9\ (10^{-9})$ cm

$\Delta\lambda$ = 0.01

e = charge on electron = 1.6×10^{-19} Coulomb

ΔF = video bandwidth = 10^7 Hz

N_b = spectral radiance of the background = 0.01 watt/cm^2-ster-micron (sunlit cloud background)

K = photocathode sensitivity = 0.002 amperes/watt (S1 photocathode at $0.9\,\mu$)

i_s = photocathode signal current

i_n = RMS shot noise current

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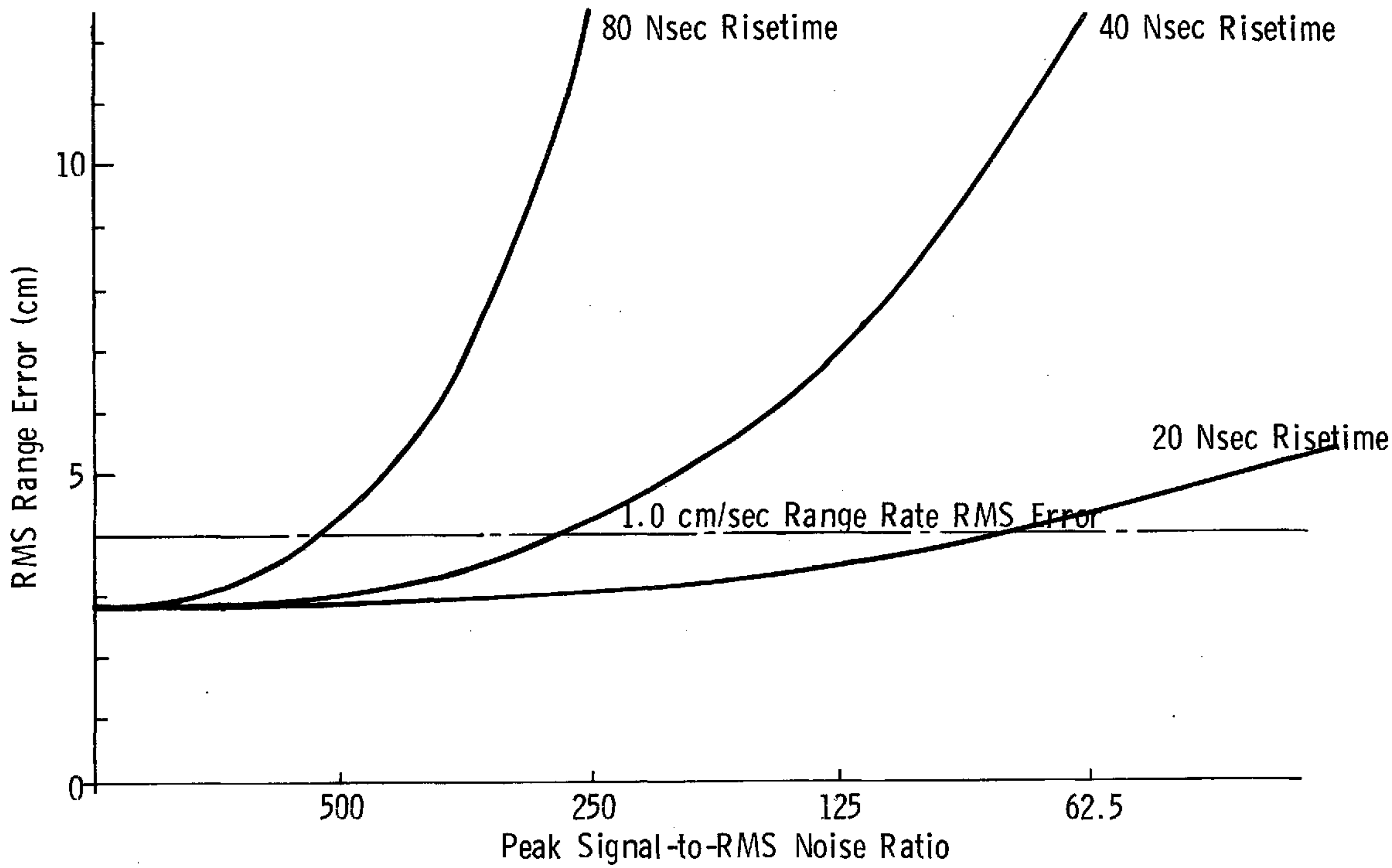
RMS RANGE ERRORS

The relative range and range-rate range error that can be expected from a laser radar is shown as a function of pulse rise-times and peak signal-to-noise ratios. Signal-to-noise ratio of 100 and rise-times of 80 N sec are easy to obtain. Range accuracy of 8 cm and range rate accuracy of 1.0 cm/sec can be obtained by a laser radar.

RMS RANGE ERROR*



* Reference NAS8-20717

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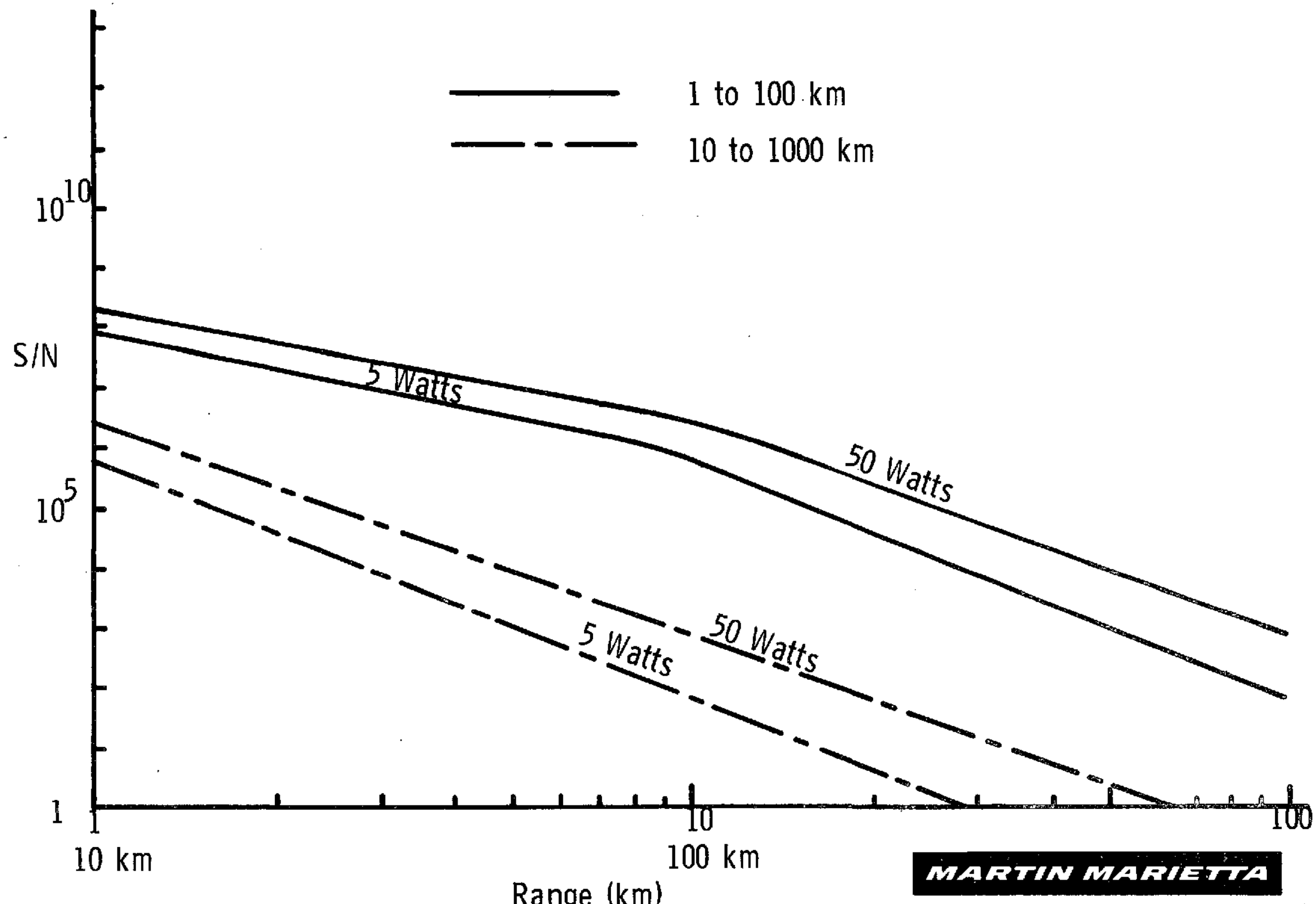
RENDEZVOUS RANGE CAPABILITY

The maximum range capability is shown for S/N ratio and output power. For a S/N ratio of 100, a maximum range of 100 Km can be achieved with 5 watts and 200 Km with 50 watts. So a 1000 Km laser radar could be easily be obtained with reasonable power. This type of radar can be used as a rendezvous radar, but a microwave radar was baselined in this study because it can be developed cheaper and its transponder can be used for multi-functions.

RENDEZVOUS RANGE CAPABILITY*



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* Reference NAS8-20717

RENDEZVOUS RADAR SPECIFICATIONS (COOPERATIVE RENDEZVOUS)

This Vugraph shows the power required on the MAV and also on the orbiter to accomplish cooperative rendezvous. The assumed microwave losses and gains are shown with the appropriate radar range equation. The MAV has a 0.51 m (20 in.) antenna and the orbiter has a 0.15 m (6 in.) receiver antenna. The specifications are based on using a pulse rendezvous radar although the baseline CW system would have about the same range, but would require additional average power.

RENDEZVOUS RADAR SPECIFICATIONS (COOPERATIVE RENDEZVOUS)

Frequency = 1 GHz
Pulse Width = 6 μsec
PRF = 256 Hz

Swerling V Model
Cooperative Target

$$P_t = \frac{(4\pi)^2 R^2 LKTBF (S/N)_{Req}}{G_t G_r \lambda^2}$$

Parameter	Contribution	Remarks
$(4\ )^2$	+ 22.0 dB	
R^2	+120.0	H_{max} = 1000 km
L	+ 10.0	
KT	-204.0	T = 290^0K
B	+ 52.2	Pulse Width = 6 μsec
F	+ 10.0	
G_t	- 19.3	MAV Ant. = 0.51 m
G_r	- 4.0	Rend. Radar Ant. = 0.15 m
λ^2	+ 10.5	λ = 0.3
$(S/N)_{Req}$	+ 8.0	
	+ 5.4 dB	

Peak Power = 3.5 Watts; Average Power = $5(10^{-3})$ Watts
Primary Power = 1.0 Watt (MAV Transponder)

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RENDEZVOUS RADAR SPECIFICATIONS (PASSIVE COOPERATIVE RENDEZVOUS)

Using a radar subsystem with the power required to skin track (non-cooperative rendezvous) the vehicle from 5 km, the maximum range capability can be increased, when the MAV is pointing at the orbiter. The rendezvous radar illumination is reflected back passively by the MAV antenna. An estimated maximum range of 35.6 km can be obtained when using a passive cooperative type of rendezvous.

RENDEZVOUS RADAR SPECIFICATIONS (PASSIVE COOPERATIVE RENDEZVOUS)

Frequency = 1 GHz

Pulse Width = 6 μsec

PRF = 256 Hz

Swerling V Model

Non-cooperative Target

$$R^4 = \frac{P_t \, G_t^{\,2} \, G_r^{\,2}}{(4\pi)^2 \, LKTBF \, (S/N)_{Req}}$$

Parameter	Contribution	Remarks
P_t	+ 52.2	P_t = 16700 Watts
G_t	+ 8.0	Rend. Radar Ant. = 0.51 m
$G_r^{\,2}$	+ 20.0	Passive MAV Ant.
$(4\pi)^2$	- 22.0	
L	- 10.0	
KT	+204.0	T = 290^{0}K
B	- 52.2	Pulse Width = 6 μsec
F	- 10.0	
$(S/N)_{Req}$	- 8.0	
	+182.0	

R_{max} = 35.6 km

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RENDEZVOUS RADAR SPECIFICATIONS (NON-COOPERATIVE RENDEZVOUS)

The rendezvous radar power requirement was determined, so the rendezvous radar would have a maximum range of 5 km when the vehicle is skin tracked (non-cooperative rendezvous). A peak power of 16.7 kilowatts would be needed or 25 watts average power.

RENDEZVOUS RADAR SPECIFICATIONS (NON-COOPERATIVE RENDEZVOUS)

Frequency = 1 GHz

Pulse Width = 6 μsec

PRF = 256 Hz

Swerling V Model

Non-cooperative Target

$$P_t = \frac{(4\pi)^3 R^4 \, LKTBF \, (S/N)_{Req}}{G_t \, G_r \, \sigma^2}$$

Parameter	Contribution	Remarks
$(4\pi)^3$	+ 33.0 dB	
R^4	+147.9	R_{max} = 5 km
L	+ 10.0	
KT	-204.0	T = 290^{0}K
B	+ 52.2	Pulse Width = 6 μsec
F	+ 10.0	
$(S/N)_{Req}$	+ 8.0	
G_t	- 4.0	Rend. Radar Ant.
G_r	- 4.0	Rend. Radar Ant.
σ	- 10.4	
χ^2	+ 10.5	
	+ 52.2 dB	

Peak Power = 16700.0 Watts

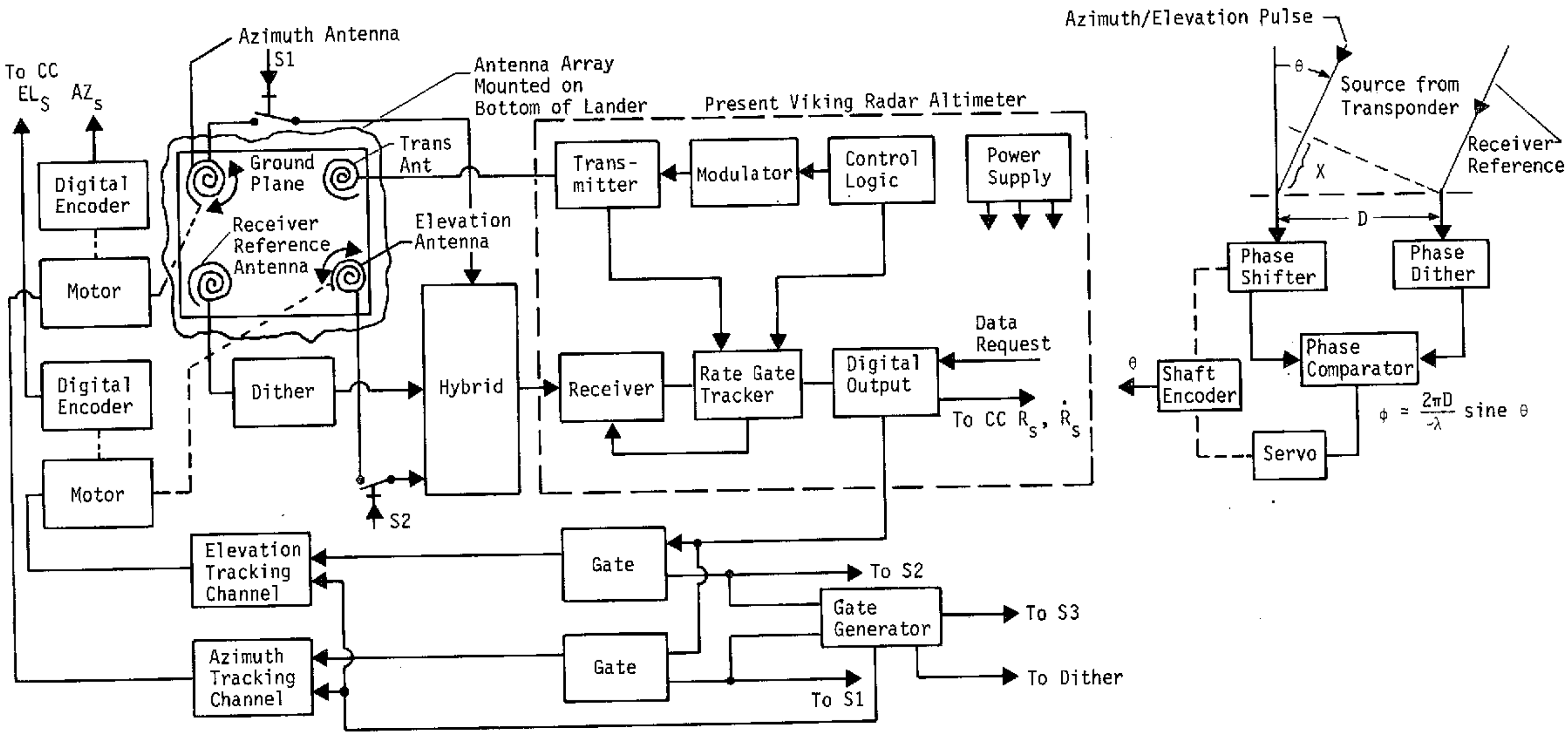
Average Power = 16700 (6) (10^{-6}) (256) = 25.0 Watts (Rendezvous Radar)

MICROWAVE PULSE RADAR

This Vugraph shows the alternate rendezvous radar, where pulse ranging and phase monopulse angle tracking is used. This implementation assumes the Viking lander radar altimeter is modified and would supply the primary components for the rendezvous radar. An antenna system and electronics for phase monopulse tracking system has to be added to the radar altimeter to implement this type of radar as shown on this Vugraph.

MICROWAVE PULSE RADAR

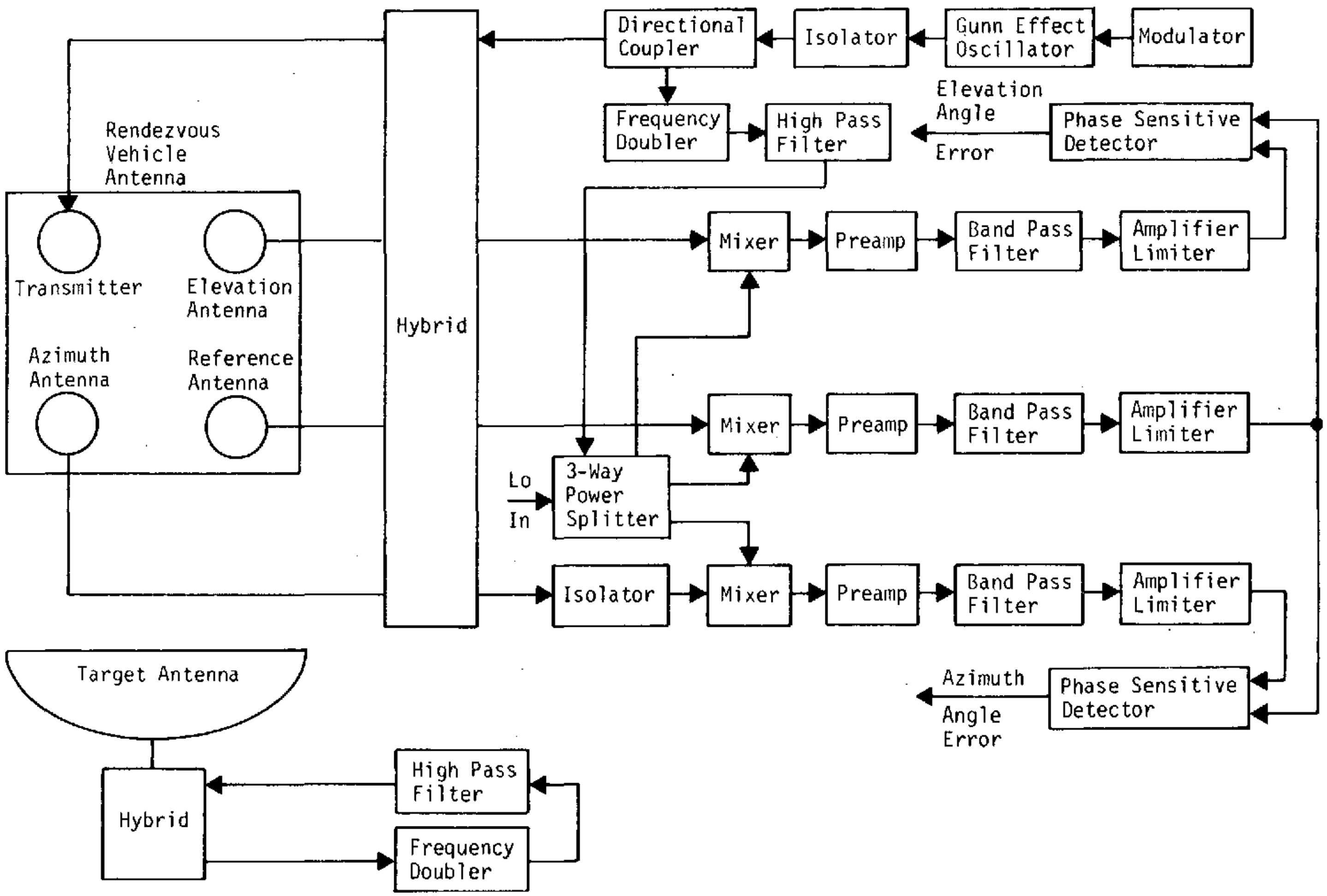
(Pulse Ranging and Phase Monopulse Angle Tracking)



FM/CW DOCKING RADAR BLOCK DIAGRAM

The alternate docking radar concept block diagram is shown. This system would weigh about 10 lbs and would utilize the same antennas as the rendezvous radar. Range and range rate can be determined very accurately with this type of system, and because these parameters can be determined very accurately the LOS angles can also be determined very precisely.

FM/CW DOCKING RADAR BLOCK DIAGRAM



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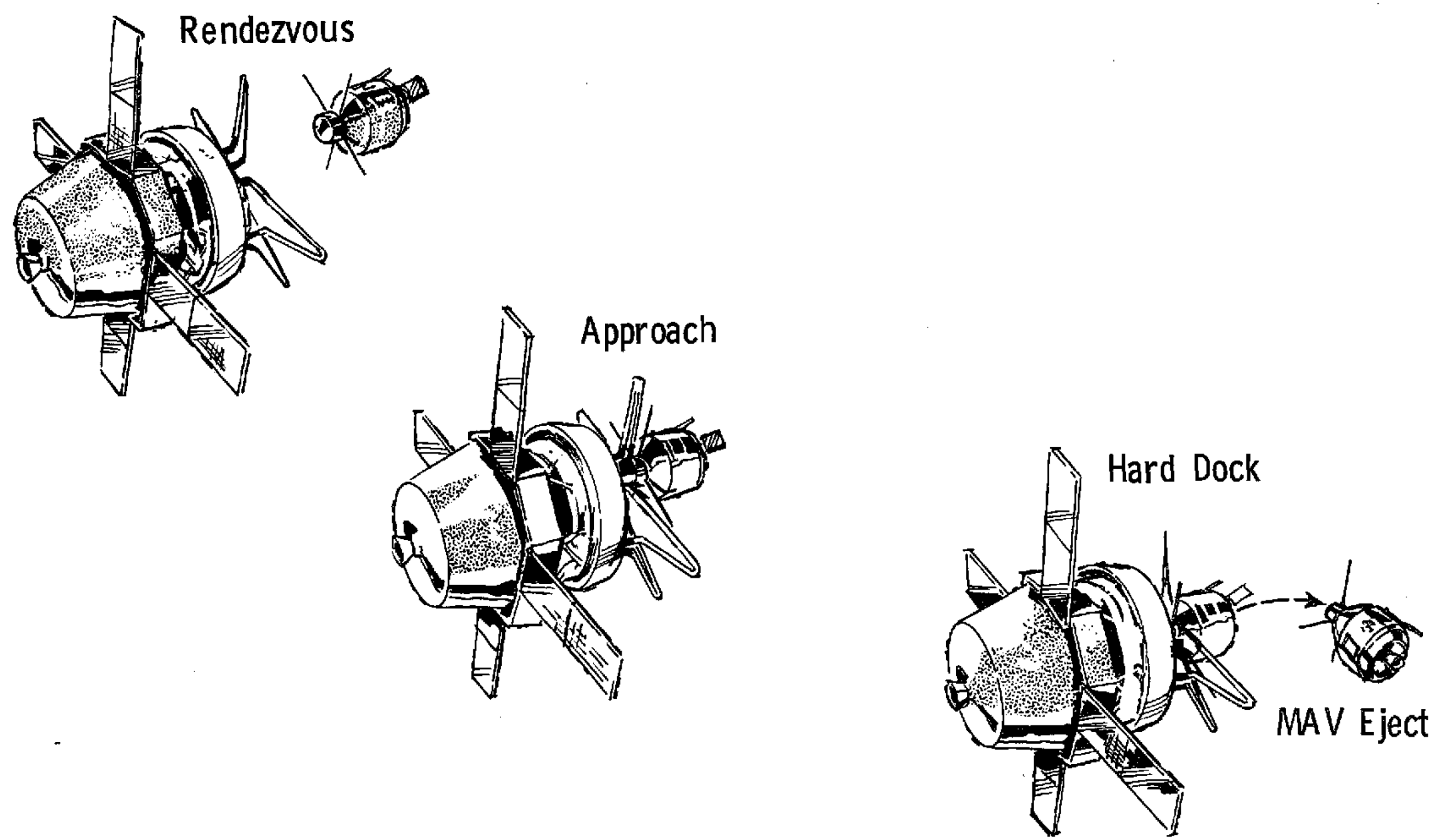
DOCKING ALIGNMENT CONCEPT



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TELECOMMUNICATIONS

J. D. Pettus

TELECOMMUNICATIONS FUNCTIONAL REQUIREMENTS

Basic telecommunications functional requirements for the Lander and MAV during surface operations, MAV in orbit and MAV during rendezvous with the orbiter are shown.

For surface operations the primary function is to provide the command and telemetry functions necessary to acquire a surface sample, transfer it to the MAV and launch the MAV in the intended orbit. A secondary function is to provide image data return from the lander.

With the MAV in orbit, a tracking and command capability from Earth is required to circularize the orbit. A further requirement of the radio subsystem is to provide pointing information to the MAV guidance system so that the vehicle may be accurately pointed to Earth during tracking and prior to maneuvers.

During rendezvous operations the telecommunications subsystems could conceivably be passive with MAV attitude (point to orbiter errors) and rendezvous transponder functions handled by radar. However, with a dual purpose system, capability for providing error signals for pointing the MAV toward the orbiter, providing cooperative range and range rate data as well as a backup maneuver command reception capability and telemetry can be assessed as a requirement for the MAV telecommunications subsystems.

TELECOMMUNICATIONS FUNCTIONAL REQUIREMENTS

MAV On Surface

- Operational Commands to Lander
- Launch Parameters and Ascent Program for MAV
- Tracking to Locate Lander/MAV
- MAV and Lander Engineering Data to Earth
- Lander Camera Data to Earth

MAV In Orbit

- 2-Way Doppler for Orbit Determination
- Earth Pointing Reference Using Radio Tracking
- Operational Commands to MAV - Orbit Trim
- MAV Telemetry to Earth

MAV During Rendezvous

- Provide Pointing Reference - (Point to Orbiter)
- Receive Commands from Orbiter
- Cooperative Ranging and Doppler Transponder

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COMPARISON OF OPTIONS FOR COMMUNICATIONS (3 AXIS STABILIZED MAV)

Surface Operations

There are several options to consider in providing communications for the Mars surface operations preceding the MAV launch.

Commands and telemetry as required for retrieving a surface sample and launching the MAV into its initial orbit could conceivably be provided using S-band equipment mounted in either the Lander or the MAV. A further consideration is use of a UHF relay for command and telemetry between the orbiter and the Lander/MAV. Preliminary findings tend toward use of S-band in the lander for surface operations to provide daily Earth contact if required as opposed to a relay link which is severely limited in communications opportunities due to the 1000 x 100,000 km initial orbit of the orbiter. A disadvantage of use of the MAV S-band equipment is the necessity of a HGA that can be gimbaled to track Earth and the need for an omni for command backup. These, even though they could be separated from the MAV in the launch attitude require extensive RF interface and impose weight penalties for the MAV.

MAV in Orbit

The need for an Earth reference for MAV attitude and the requirement to determine the MAV orbit from Earth Tracking leave little option for use of other than an S-band MAV/Earth communications capability. For a three axis stabilized MAV a monopulse type angle sensor and a typical DSN two-way Doppler, command and telemetry system appear to best fulfill the needs.

MAV in Rendezvous

During rendezvous of MAV and orbiter the MAV S-band subsystem used for MAV/Earth communications and pointing error data during MAV orbit adjust could serve the same functions in interfacing with the orbiter as with the DSN by adding a ranging turnaround capability and providing a means for operating at appropriate frequencies. The alternative is to provide a separate rendezvous and docking subsystem.

An angle tracking dual ratio S-band transponder has tentatively been selected to provide pointing, communication and tracking capabilities when interfacing with either the DSN or the orbiter in the [illegible] orbit and MAV rendezvous modes.

COMPARISON OF OPTIONS FOR COMMUNICATIONS (3 AXIS STABILIZED MAV)

MAV on Surface - Options for Surface Operations

- DSN/Lander S-Band
- DSN/MAV S-Band
- Lander/Orbiter UHF

MAV In Orbit - Options for MAV Orbital Operations

- DSN/MAV S-Band - (Command, Telemetry - 2-Way Doppler, Monopulse)
- Other Than Monopulse

MAV In Rendezvous Mode - Options

- S-Band Multipurpose Transponder (Shift to Rendezvous Frequencies)
- Separate Rendezvous Transponder
  - CW with Tones
  - Pulse

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PROPOSED LANDER TELECOMMUNICATIONS BLOCK DIAGRAM

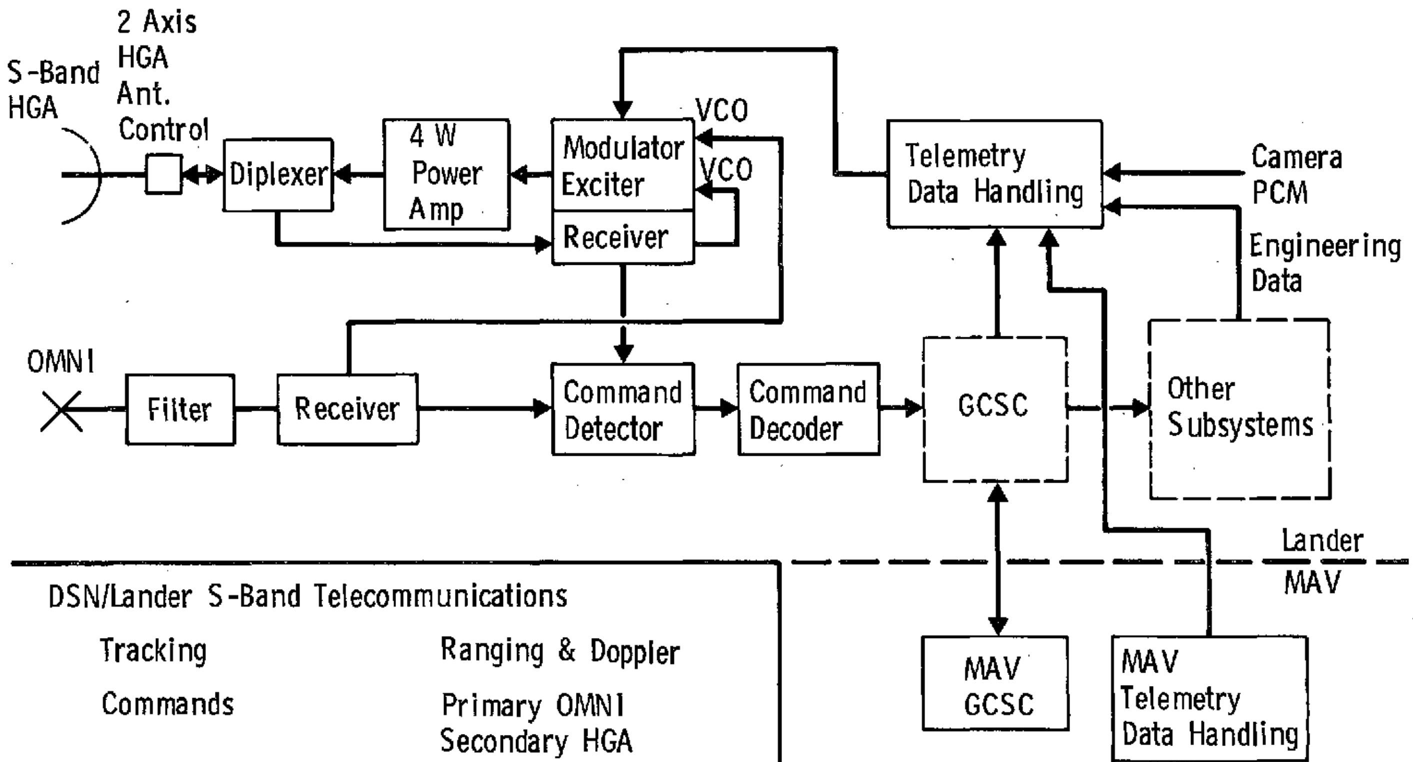
All surface communications with Earth are carried out through a lander S-band system using "light weight" MAV components where possible such as the 4 W solid-state power amplifier, modulator/exciter command detector and decoder. The Viking '75 high and low gain S-band antennas and antenna drive mechanisms are retained. No UHF links to the orbiter are included. Control of the high gain antenna (HGA) pointing is accomplished using the Viking Guidance Control and Sequencing Computer (GCSC).

Two single channel telemetry rates are provided, 8 1/3 bps uncoded for engineering and 250 bps (block coded) for video data. Both rates are for the HGA. The omni provides a receive only primary command capability. Secondary command and ranging are via the HGA.

MAV engineering data for transfer to Earth and Earth command data for updating the MAV computer are transferred between the lander and MAV via umbilical prior to launch of the MAV using a digital interface.

More detailed interface and functional studies are required for further definition. Mounting of the S-band antennas must be such that a communications link with Earth is highly probable after the MAV is raised to the launch position.

PROPOSED LANDER TELECOMMUNICATIONS BLOCK DIAGRAM



DSN/Lander S-Band Telecommunications

Tracking	Ranging & Doppler
Commands	Primary OMNI Secondary HGA
Telemetry	250 bps Camera or 8-1/3 bps Engineering
Subsystem Weight	29.5 lb
Subsystem Power	
Receiving Standby	5.9 W
Full Power	32.4 W

PROPOSED MAV TELECOMMUNICATIONS BLOCK DIAGRAM

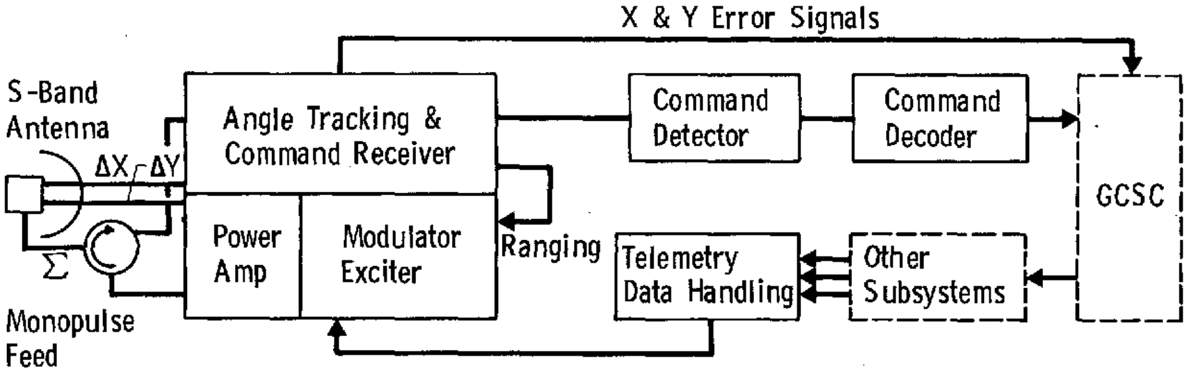
The MAV telecommunications subsystem consists of a monopulse fed 18 dB gain antenna, an angle tracking dual ratio transponder, command detector, command decoder and telemetry data handling circuity packaged in an integrated case. Angle tracking errors are obtained by a cassegrain monopulse feed and frequency sharing of a common sum channel receiver by generating error channel sideband signals and frequency multiplexing the sum and error signals.

Telemetry and command are DSN compatible PSK/PM with two-way coherent Doppler. Turnaround ratio is 240/221 for DSN operation and tentatively 220/239 for orbiter interfacing. Turnaround ranging is intended only for the MAV/orbiter rendezvous. The command subsystem is single channel using sinewave subcarrier. Telemetry is single channel squarewave. The 4 watt MIC power amplifier is sized for MSC 3005 transistors and 20 volts D.C. input.

The Guidance Computer and Sequencer (GCSC) provides the power turn-on control for the telecommunications except that an uplink receive signal enables turn on of the command detector and decoder. Low power designs are contemplated for all units.

Selection of monopulse and a single receiver channel type angle tracking receiver to obtain attitude reference is tentative. Final selection of monopulse and 1, 2 or 3 receiver channels requires additional analysis to weigh the tradeoffs and performance attainable.

PROPOSED MAV TELECOMMUNICATIONS BLOCK DIAGRAM



Subsystem
- Weight: 4.7 lb Uncased
- Max Power: 21.3 W
- Min Power: 3.5 W

Characteristics	DSN/MAV Link	Orbiter/MAV Link
Tracking	2-Way Doppler	Doppler, Ranging & Angle
MAV Attitude (Pointing)	S-Band Monopulse Feed & Single Channel Rec.	Same
Commands	Single Channel Subcarrier	Same
Telemetry	8-1/3 bps	Same
Transmitter Power Amp	4 Watts	4 Watts/150 mW
MAV Transmit Freq	2292.03 MHz	2101.03 MHz
MAV Receiver Freq	2110.58 MHz	2282.48 MHz

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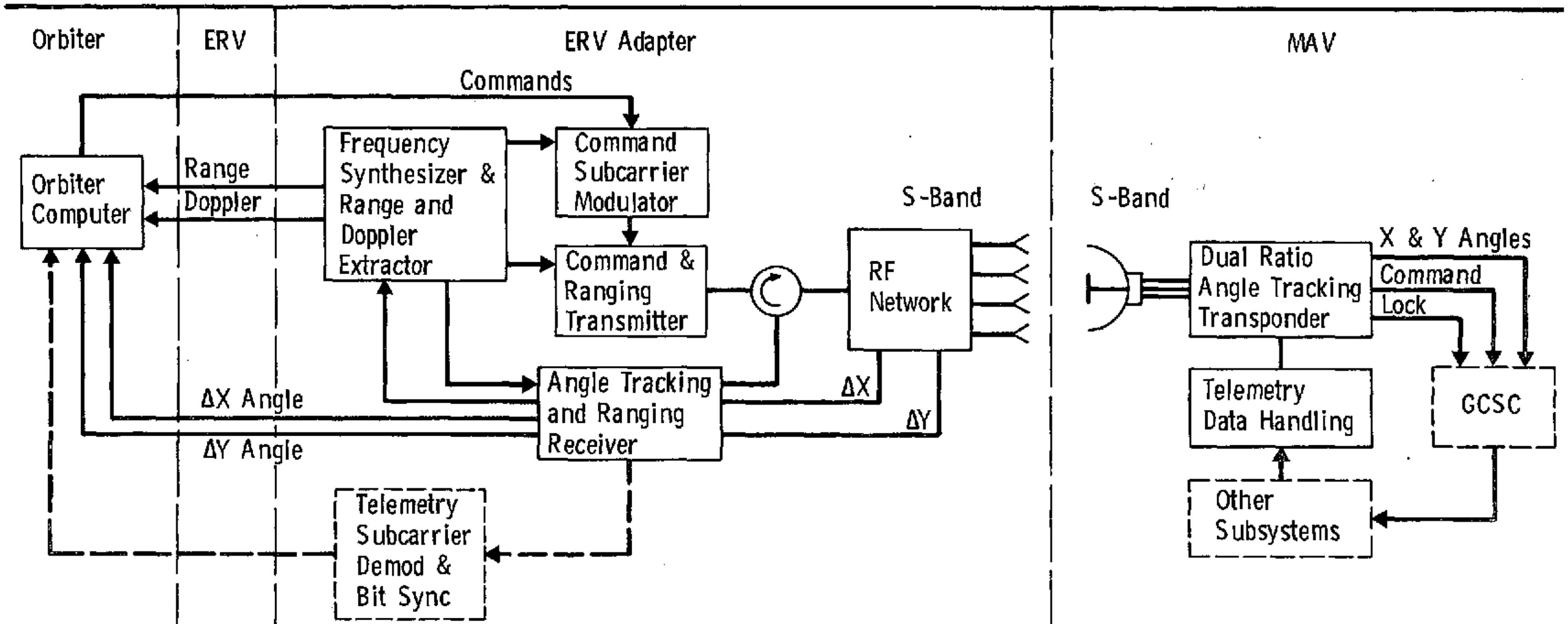
PROPOSED ORBITER/MAV COMMUNICATIONS

In the normal rendezvous and docking mode the MAV S-band transponder provides turn around for a coherent ranging signal, demodulates a command subcarrier and combines a PSK modulated subcarrier with the ranging for transmission to the orbiter. Commands from the orbiter will be required only in event the orbiter cannot maneuver for rendezvous. In this case the MAV could be commanded to start or stop thrust. Thus command is back up only. Telemetry from MAV to Earth via the orbiter is desirable.

The orbiter must perform the rendezvous and docking maneuvering once it is in the desired orbit. To accomplish this an S-band CW range, range rate and angle tracking system is provided using a 14 dB three channel monopulse antenna and receiver system, a 100 MW transmitter and coherently generated range tones. The highest frequency tone is 163.84 kHz which provides a resolution of $\sim$10 meters. Three additional tones are used for resolving range ambiguity for the maximum required range of 250 km. These tones are modulated onto the highest frequency tone prior to transmission by the orbiter and demodulated when received from the MAV turnaround.

The presently conceived interface for the orbiter equipment is to mount it near the sample transfer cone (using the cone to support surface wave antenna elements) and carry power and digital signals through connector interfaces between the ERV and the orbiter main body. After docking and transfer, the cone and S-band equipment may be jettisoned.

PROPOSED ORBITER/MAV COMMUNICATIONS



Two-Way Signal Acquisition	Command Backup (Orb to MAV)
Orbiter Initiate Transmit Sweep MAV Acquire Then Orb Acquire MAV	Thrust Start/Stop Return to Earth Pointing & Control
Ranging Doppler & Angles	**Desirable Telemetry at Additional Cost**
Turn Around Ranging Tones Monopulse Angle Tracking	MAV Engineering to Earth Via Orbiter

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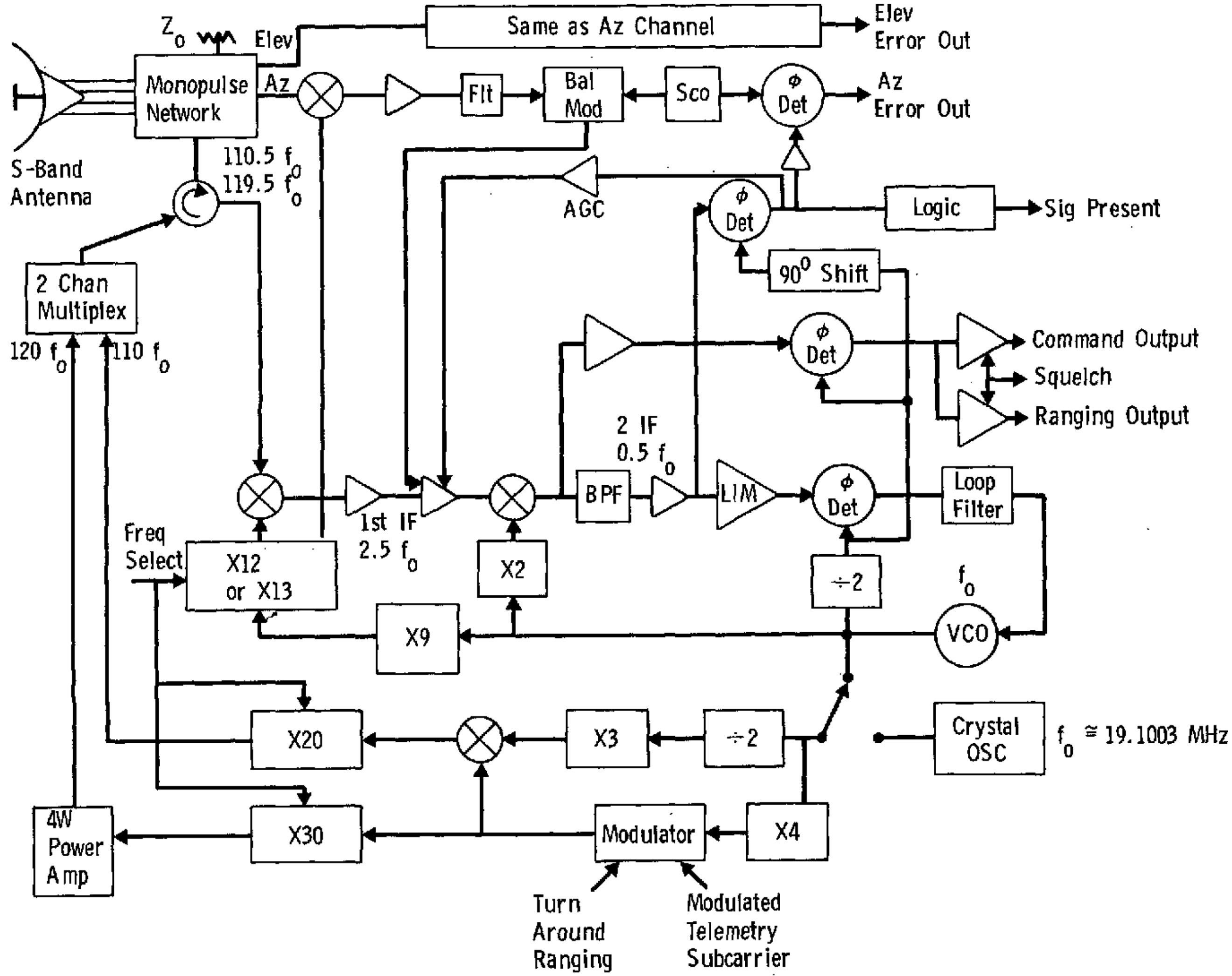
MAV ANGLE TRACKING DUAL RATIO TRANSPONDER

Single IF chain angle tracking is chosen over a conventional three channel receiver since it is lighter and contains much less equipment. The simplification occurs by replacing the error signal IF chains by a crystal filter, balanced modulator, and low frequency oscillator. Tradeoffs indicate for this system a reduction in size, weight, and power, and an increase in reliability. The price paid for this improvement is possibly 3 dB decrease in S/N ratio and reduction in sensitivity due to phase shifts. Full impact on performance is being evaluated.

The error signals are converted to the 1.I.F. with mixers identical to those in the sum channel. Each error signal is then modulated with a distinct tone in the balanced modulator producing sidebands whose amplitude is proportional to the amplitude of the error signals. The error sidebands which are outside the normal modulation sideband of the reference channel are added to the sum channel. This composite signal after conversion to the 2.I.F. passes through a multiple crystal filter which places a narrow bandpass about the carrier as well as a narrow bandpass about one of the sidebands or each error signal. The command and ranging signals are stripped off before these multiple filters. After amplification the error signals are detected in coherent amplitude detectors, which are basically phase detectors with reference signals which are in phase with the carrier signal. The amplitude and phase of the error tones are then determined.

The dual-ratio transponder utilizes the sum channel from the monopulse antenna. The coherence ratio is 240/221 for the standard DSIF link and 220/239 for the rendezvous link with the orbiter. For a 220/239 transponder ratio, the transponder receives at 119.5 f_o and transmits at 110 f_o where f_o is the VCO frequency. For a 240/221 transponder ratio, the transponder receives at 110.5 f_o and transmits at 120 f_o. A system of mixing and multiplication is employed to achieve these ratios, and the appropriate chain is selected for either the DSN or rendezvous function.

MAV ANGLE TRACKING DUAL RATIO TRANSPONDER



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ORBITER RENDEZVOUS AND COMMAND SYSTEM

A multitone PM/CW rendezvous and command system is employed to acquire, track and rendezvous with the MAV vehicle. This system is simple, small, lightweight and requires minimum power.

A phase comparison monopulse system utilizing four corrugated surface wave antennas located in the sample transfer guide cone and a monopulse beam forming network is employed to provide angle tracking in both the azimuth and elevation planes.

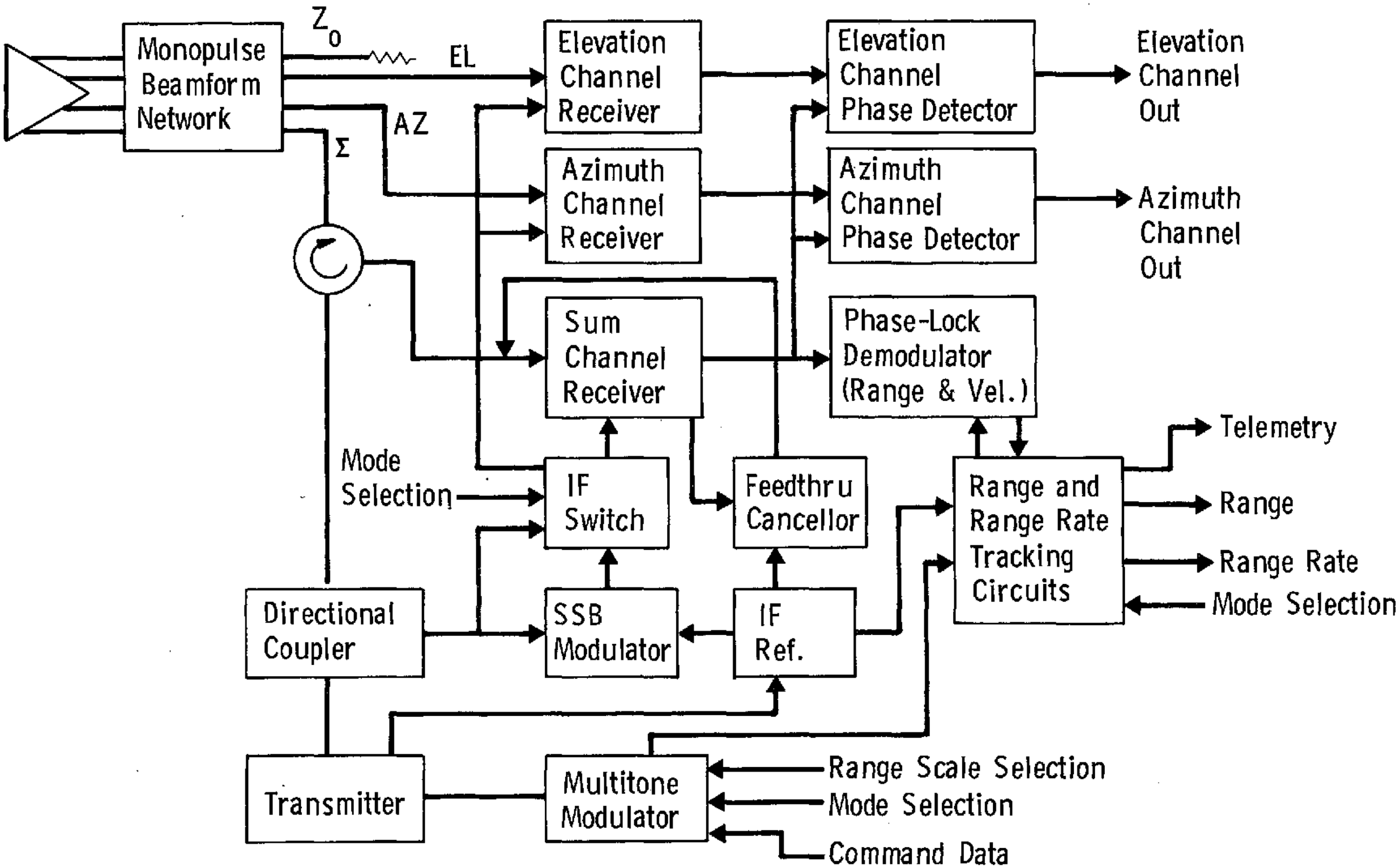
The location of the four antennas along the guide cone permits the sample canister to be transferred beyond the effective aperture plane of the array. This allows the command system to function even after transfer of the sample canister.

Dual-mode operation of the rendezvous system is provided. In the normal transponder mode the transmitter operates at 2282.48 mhz and the MAV beacon transponder translates this frequency by the 1.I.F. frequency of the rendezvous receiver. This results in a transponder frequency of 2101.3 mhz. In the non-cooperative, skin track mode (failure mode) a single sideband modulator is employed to offset a sample of the transmitted signal by the I.F. for use as the local oscillator signal. Feed-through cancelling circuits are employed in this mode.

The multitone generator produces the range tone frequencies and reference pulses. The command data are added to the modulated subcarrier to obtain a composite modulation signal, which phase modulates the solid state S-band transmitter.

The rendezvous system receiver demodulates the transponded signal, and a phase locked loop is employed to lock onto the retransmitted carrier. This loop recovers the range tones and doppler information used to obtain range and range rate.

ORBITER RENDEZVOUS AND COMMAND SYSTEM -- SIMPLIFIED BLOCK DIAGRAM



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LANDER TO EARTH TELEMETRY - SURFACE OPERATIONS

The highest telemetry data rate proposed for the Lander to Earth link is 250 bps using the Lander high gain antenna. The vugraph shows the major design control table parameters for this link and illustrates the fact that a 4 watt S-band power amplifier and the Viking Lander HGA are adequate for this data rate.

Although not shown, an 8 1/3 bit per second data rate and turn around ranging can be accomplished simultaneously.

LANDER TO EARTH TELEMETRY - SURFACE OPERATIONS

No.	Parameter	Nominal Value	Adverse Tolerance	Notes
1	Total Transmitting Power (dBm)	+ 36.0	0.6	4 Watts
2	Transmitting Circuit Loss (dB)	- 0.9	0.2	
3	Transmitting Antenna Gain (dB) (Viking Lander HGA)	+ 21.1	0.3	1 dB Pointing
4	Communications Range Loss (dB) 2292 MHz	-267.9	0	257×10^6 km
5	Atmospheric Absorption & Defocusing Losses (dB)	0	0	
6	Polarization Loss (dB)	- 0.1	0	
7	Multipath and Other Losses (dB)	0	0	
8	Receiving Antenna Gain (dB)	+ 61.4	0.4	64 Meter Net
9	Receiving Circuit Loss (dB)	0	0	
10	Net Loss (dB) (2+3+4+5+6+7+8+9)	-186.4	0.9	
11	Total Received Power (dBm) (1+10)	-150.4	1.5	
12	Receiver Noise Spectral Density (dBm/Hz)	-184.2	0.5	25^o Elev.
13	Total Received Power/N_o (dBm·Hz) (11-12)	+ 33.8	2.0	
	Carrier Tracking			θ = 1.16 Rad.
14	Carrier Power/Total (dB)	- 8.0	2.7	
15	Additional Carrier Losses (dB)	- 0.1	0	
16	Threshold Tracking Bandwidth - $2B_{LO}$ (dB)	+ 10.8	0	
17	Threshold SNR (dB)	+ 10.0	0	
18	Performance Margin (dB) (13+14+15-16-17)	+ 4.9	4.7	
	Data Channel			θ = 1.16 Rad.
19	Data Power/Total (dB)	- 0.8	0.5	
20	Additional Data Channel Losses (dB)	- 2.0	0.3	Estimated
21	Data Bit Rate - bps (dB)	+ 24.0	0	250 bps
22	Threshold Energy Per Data Bit - E_b/N_o (dB)	+ 3.0	0	10^{-2}W
23	Performance Margin (dB) (13+19+20-21-22)	+ 4.0	2.8	

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EARTH TO LANDER COMMAND - SURFACE OPERATIONS & LOW GAIN ANTENNAS

Initial or primary command to the Lander during surface operations is via the Lander Omni and the 64 meter DSN net. The vugraph shows that there is ample margin for uplink command to the Lander via the omni antenna to activate a 2-way link using the Lander high gain antenna.

EARTH TO LANDER COMMAND - SURFACE OPERATIONS AND LOW GAIN ANTENNA

No.	Parameter	Nominal Value	Adverse Tolerance	Notes
1	Total Transmitting Power (dBm)	+ 70.0	0	10 kW
2	Transmitting Circuit Loss (dB)	0	0	
3	Transmitting Antenna Gain (dB)	+ 60.4	0.7	64 Meter Net
4	Communications Range Loss (dB) (2110.58 MHz)	-267.1	0	257×10^6 km
5	Atmospheric Absorption & Defocusing Losses (dB)	0	0	
6	Polarization Loss (dB)	- 0.4	0	
7	Multipath and Other Losses (dB)	0	0	
8	Receiving Antenna Gain (dB)	+ 1.3	0.5	
9	Receiving Circuit Loss (dB)	- 1.3	0.2	-3.2 dB Pointing
10	Net Loss (dB) (2+3+4+5+6+7+8+9)	-207.2	1.4	
11	Total Received Power (dBm) (1+10)	-137.2	1.4	
12	Receiver Noise Spectral Density (dBm/Hz)	-167.5	0.8	1300° K
13	Total Received Power/N_0 (dBm·Hz) (11-12)	+ 30.3	2.2	
	Carrier Tracking			
14	Carrier Power/Total (dB)	- 2.5	0.2	
15	Additional Carrier Losses (dB)	0	0	
16	Threshold Tracking Bandwidth - $2B_{LO}$ (dB)	+ 12.6	0.5	
17	Threshold SNR (dB)	+ 10.0	0	
18	Performance Margin (dB) (13+14+15-16-17)	+ 5.2	2.9	
	Data Channel			
19	Data Power/Total (dB)	- 4.0	0.2	
20	Additional Data Channel Losses (dB)	- 1.5	0.2	
21	Symbol Rate - SPS (dB)	+ 6.0	0	4 SPS
22	Threshold Energy Per Symbol - E_S/N_o (dB)	+ 11.5	1.0	10^{-5}
23	Performance Margin (dB) (13+19+20-21-22)	+ 7.3	3.6	

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MAV TO EARTH COMMUNICATIONS - MAV IN ORBIT

The design control table shows the major prarmeters in a MAV to Earth telemetry link. Adequate margin is provided for an 8 1/3 bit per second data rate using a 4 watt transmitter and an 18 db gain antenna when the MAV antenna is pointing up to 10 degrees off the Earth/MAV line.

MAV TO EARTH COMMUNICATIONS - MAV IN ORBIT

No.	Parameter	Nominal Value	Adverse Tolerance	Notes
1	Total Transmitting Power (dBm)	+ 36.0	0.4	4 Watts
2	Transmitting Circuit Loss (dB)	- 1.5	0.2	
3	Transmitting Antenna Gain (dB) 2292 MHz	+ 15.0	0.5	18 dB on Axis
4	Communications Range Loss (dB)	-267.9	0.0	257×10^6 km
5	Atmospheric Absorption & Defocusing Losses (dB)	0	0	
6	Polarization Loss (dB)	- 0.3	0.3	
7	Multipath and Other Losses (dB)	0	0	
8	Receiving Antenna Gain (dB)	+ 61.5	0.4	64 Meter
9	Receiving Circuit Loss (dB)	0	0	
10	Net Loss (dB) (2+3+4+5+6+7+8+9)	-193.2	1.2	
11	Total Received Power (dBm) (1+10)	-157.2	1.6	
12	Receiver Noise Spectral Density (dBm/Hz)	-184.2	0.5	25° Elev.
13	Total Received Power/N_0 (dBm·Hz) (11-12)	+ 27.0	2.1	
	Carrier Tracking			0.613 Rad.
14	Carrier Power/Total (dB)	- 1.7	0.4	
15	Additional Carrier Losses (dB)	0.1	0	
16	Threshold Tracking Bandwidth - $2B_{LO}$ (dB)	+ 10.8	0	
17	Threshold SNR (dB)	+ 10.0	0	
18	Performance Margin (dB) (13+14+15-16-17)	+ 4.4	2.5	
	Data Channel			
19	Data Power/Total (dB)	- 4.8	0.8	
20	Additional Data Channel Losses (dB)	- 2.9	0.3	
21	Data Bit Rate - bps (dB)	+ 9.2	0	8-1/3 bps
22	Threshold Energy Per Data Bit - E_b/N_o (dB)	+ 5.2	0	Uncoded
23	Performance Margin (dB) (13+19+20-21-22)	+ 4.9	3.2	

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EARTH TO MAV COMMAND - MAV IN ORBIT

The design control parameters for an Earth to MAV command link are shown for the 64 meter net and a 10 KW transmitter. Over 14 db excess margin is available for establishing an uplink even when the MAV antenna is 10° off Earth pointing.

No omni capability has been provided because of weight constraints; however, a monopulse system has been provided to allow the MAV to sense Earth direction and correct the vehicle attitude so as to point the antenna to Earth.

EARTH TO MAV COMMAND - MAV IN ORBIT

No.	Parameter	Nominal Value	Adverse Tolerance	Notes
1	Total Transmitting Power (dBm)	+ 70.0	0	10 kW
2	Transmitting Cricuit Loss (dB)	0	0	
3	Transmitting Antenna Gain (dB)	+ 60.4	0.7	64 Meter Net
4	Communications Range Loss (dB) 2110.58 MHz	-267.1	0	257×10^6 km
5	Atmospheric Absorption & Defocusing Losses (dB)	0	0	
6	Polarization Loss (dB)	- 0.4	0	
7	Multipath and Other Losses (dB)	0	0	
8	Receiving Antenna Gain (dB)	+ 14.3	0.5	-3 dB Pointing
9	Receiving Circuit Loss (dB)	- 2.5	0.3	
10	Net Loss (dB) (2+3+4+5+6+7+8+9)	-195.3	1.5	
11	Total Received Power (dBm) (1+10)	-125.3	1.5	
12	Receiver Noise Spectral Density - (dBm/Hz)	-167.5	0.8	1300°K
13	Total Received Power/N_0 (dBm·Hz) (11-12)	+ 42.6	2.3	
	Carrier Tracking			
14	Carrier Power/Total (dB)	- 2.5	0.2	
15	Additional Carrier Losses (dB)	0	0	
16	Threshold Tracking Bandwidth - $2B_{LO}$ (dB)	+ 12.6	0.5	18 Hz
17	Threshold SNR (dB)	+ 10.0	0	
18	Performance Margin (dB) (13+14+15-16-17)	+ 17.5	3.0	
	Data Channel			
19	Data Power/Total (dB)	- 4.0	0.2	
20	Additional Data Channel Losses (dB)	- 1.5	0.2	
21	Symbol Rate - SPS (dB)	+ 6.0	0	4 SPS
22	Threshold Energy Per Symbol - E_s/N_0 (dB)	+ 11.5	1.0	10^{-5}
23	Performance Margin (dB) (13+19+20-21-22)	+ 19.6	3.7	

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SYSTEMS SUMMARY

Vehicle Configurations - N. M. Phillips
Mass Properties - W. D. VanArnam
Aerodynamics - G. L. Cahen
Propulsion - R. F. Fearn and C. E. Lynch
Power - A. A. Sorensen
Thermal Control - T. Buna

J. R. Mellin

URDMO ORBITER MASS DERIVATION

Viking orbiter modifications include removal of science associated items including all orbiter science, the scan platform, and the data storage system. Values shown for these items include associated structure, cabling, insulation, and articulation mechanisms. In addition, the cold gas RCS system is replaced with a monopropellant maneuver/RCS system. This change is made to provide a low thrust system for rendezvous and docking. Estimated mass of this system is 34 kg including a 10% contingency and has 13.6 kg of propellant in a single sphere approximately 35 cm in diameter. The sphere will fit in the location now occupied by one of the VO'75 RCS nitrogen bottles.

VO'75 propulsion system is stretched 20% to provide 2200 kilometer orbit insertion.

For this configuration, the rendezvous radar is assumed part of the ERV.

URDMO ORBITER MASS DERIVATION

	kg	lb
Viking Orbiter (Dry)	917.88	2023.6
Remove Science	- 81.19	-179.0
Remove Scan Platform	- 32.60	- 72.0
Remove Data Storage	- 31.30	- 69.0
Remove Cold Gas RCS	- 44.45	- 98.0
Add Combined Maneuver/RCS	+ 34.02	+ 75.0
URDMO Orbiter (Assuming No Propulsion Change)	762.30	1680.6
Propulsion Change	35.33	77.9
Propellant Required	1710.22	3770.4
Total URDMO Orbiter	2507.85	5528.9

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URDMO LANDER MASS DERIVATION

Mass effect of changes to the Viking Lander to provide for carrying the MAV to a landing site on Mars are shown. Of these the majority are for the purpose of reducing mass, however, the change to the regulated pressure system is made to provide a higher landed mass capability. This change requires a new landing propellant and pressurization system replacing the current Viking blowdown system with a pressure regulated system. The system shown here is based upon tankage sized for 75.3 kg of propellant. Only 70.3 kg is required for the present configuration.

Other changes to lander bus cover those changes made to parts of the lander system other than the final landing stage. These include raising the parachute to provide space for mounting the MAV and increased aeroshell and heatshield required for direct entry.

URDMO LANDER MASS DERIVATION

	kg	(lb)
Viking Lander (Landed 2/19/74)	594.2	1310.0
Remove UHF	- 5.85	- 12.9
Reduce RTG Size	-22.54	- 49.7
Remove One Battery (1/2 Package)	-11.47	- 25.3
Remove Data Storage	-13.83	- 30.5
Modify Thermal System	- 5.35	- 11.8
Remove Science (except one camera & soil sampler)	-60.55	-133.5
Add Regulated Pressure System	+ 1.45	+ 3.2
Modify Telemetry	- 6.58	- 14.5
Modify S-Band to MAV Components	-15.15	- 33.4
Remove Cabling	- 9.98	- 22.0
Add MAV	+288.93	+637.0
Add MAV Launcher (incl. Thermal Protection)	+ 41.05	+ 90.5
URDMO Landed	774.33	1707.1
Other Changes to Lander Bus		
Raise Parachute 23.3 inches	+ 8.7	+ 19.1
Aeroshell Structure (Direct Entry)	+49.99	+110.2
Heat Shield (Direct Entry)	+13.61	+ 30.0
Remove Science	- 7.80	- 17.2

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290 KILOGRAM MAV MASS SUMMARY

This summary presents stage mass data for the total MAV vehicle. Solid motors are estimated on the basis of a .88 mass fraction. Stage III reaction control propellant required to correct for thrust misalignment has been accounted for when sizing lower stages on the basis of .9 kg and 7 m/sec during 1st stage burn and .3 kg and 4.6 m/sec during 2nd stage burn.

290 KILOGRAM MAV MASS SUMMARY

	kg	lb	kg	lb
Stage III				
Structure & Mechanism	8.85	19.5		
Equipment	9.39	20.7		
Propellant Inert (incl. residual)	11.29	24.9		
Contingency 10%	2.90	6.4		
Propellant	8.30	18.3		
Total Step 3			40.73	89.8
Sample			1.00	2.2
Stage III at Liftoff			41.73	92.0
Stage II				
Skirt	3.95	8.7		
Propulsion Inert	11.11	24.5		
Propellant	81.55	179.8		
Total Step 2			96.61	213.0
Stage II at Liftoff			138.34	305.0
Stage I				
Skirt	5.67	12.5		
Propulsion Inert	17.51	38.6		
Propellant	128.41	283.1		
Total Step 1			151.59	334.2
Stage I at Liftoff			289.93	639.2

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MAV STAGE III MASS STATEMENT (1)

Detail mass estimates shown are based upon projecting technology and packaging to incorporate maximum use of advanced indigrated circuitry such as hybridized CMOS. Mass for electronic components is uncased, and all subsystems are packaged in three boxes which could be integral with the body structure providing minimum mass.

MAV STAGE III MASS STATEMENT (1)

		kg (lb)	
Structure & Mechanism		8.85	(19.5)
Body (incl. Electronic Chassis & Insulation)	5.35		
Antenna Dish & Cone	.91		
Sample Canister & Mechanism	1.91		
Solar Panel & Mechanism	.68		
Radio Frequency System		1.59	(3.5)
Telemetry Unit		.41	(.9)
Guidance & Control		3.27	(7.2)
Sensors	1.68		
Electronics	1.59		
Power		3.44	(7.6)
Solar Array .11 m^2	.36		
Battery Ni-H_2 50 Whr	.68		
Control, Charger, & Reg.	2.40		
Cabling		.68	(1.5)
Total Non-propulsion		18.24	(40.2)

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MAV STAGE III MASS STATEMENT (2)

The detail mass estimate continues on this vugraph showing the propulsion system. Thruster mass is based upon developed Hamilton Standard units. Three sizes are used four 54 N units firing aft, four .45 N units firing forward and four 1.8 N units for roll control. Propellant is carried in two spherical tanks with a three to one blowdown ratio. Mass shown for the propulsion system covers all propulsion dependent mass, including structure which is frequently not included when calculating mass fraction. Mass fraction values for this system are further confused because 1.81 kg of propellant is used for RCS. Therefore, mass fraction based upon total propellant including 10% hardware congingency is .40 and based upon Δ V propellant only is .34.

Total mass values shown are for total stage including mass from the previous vugraph.

MAV STAGE III MASS STATEMENT (2)

		kg	(lb)
Propulsion		11.29	(24.9)
Thrusters	3.99		
Valves & Piping	1.90		
Tanks	2.00		
Supports	2.81		
Residuals (Gas, .1 kg; Prop., .4 kg)	.50		
Contingency 10%		2.90	(6.4)
Total Stage III Burnout		32.43	(71.5)
Propellant		8.3	(18.3)
Delta V	6.49		
RCS	1.81		
Total Stage III		40.73	(89.8)

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LANDER C.G. AND INERTIA COMPARISON

This vugraph presents mass properties for the two critical descent conditions, entry and landed. Both Viking and current configuration figures are shown for comparison.

LANDER C.G. AND INERTIA COMPARISON

	Viking	URDMO 290 kg MAV
Entry Condition		
Weight, kg (lb)	943.83 (2080.80)	1202.50 (2651.00)
C.G. X, cm (in)	-90.90 (-35.80)	-104.90 (-41.30)
C.G. Y, cm (in)	.46 (.18)	-.18 (-.07)
C.G. Z, cm (in)	-5.08 (-2.00)	-2.23 (-.88)
Inertia I_x, kg m^2 (slug ft^2)	751.00 (554.00)	1086.00 (801.00)
Inertia I_y, kg m^2 (slug ft^2)	418.00 (308.00)	648.00 (478.00)
Inertia I_z, kg m^2 (slug ft^2)	502.00 (370.00)	681.00 (502.00)
Landed Condition		
Weight	587.72 (1295.70)	774.30 (1707.10)
C.G. X, cm (in)	-91.95 (-36.20)	-110.70 (-43.60)
C.G. Y, cm (in)	-.36 (-.14)	-.81 (-.32)
C.G. Z, cm (in)	-5.28 (-2.08)	-2.57 (-1.01)
Inertia I_x, kg m^2 (slug ft^2)	317.00 (234.00)	556.00 (410.00)
Inertia I_y, kg m^2 (slug ft^2)	165.00 (122.00)	266.00 (196.00)
Inertia I_z, kg m^2 (slug ft^2)	214.00 (158.00)	240.00 (177.00)

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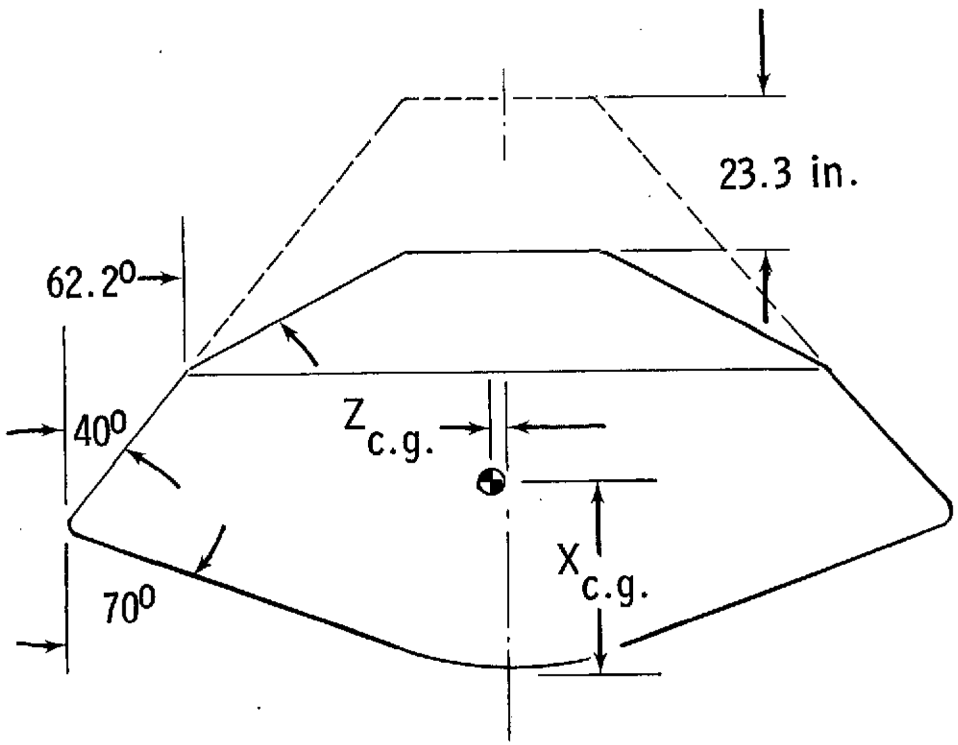
ENTRY VEHICLE GEOMETRY AND MASS PROPERTIES COMPARED TO VIKING

Several differences between the URDMO vehicle and Viking that have an influence on their aerodynamic behavior are indicated in the facing page. The mass property values given in the table were used in computing aerodynamic stability relations. These values have changed somewhat, see the Lander Table of C.G. locations and inertias for current values, but not sufficiently to invalidate the aerodynamic calculations. Based on tests of various after body shapes it is believed that extending the after body as indicated will not cause the pitch damping coefficients to vary outside the tolerance band used in the Viking design. Based on these Viking aero-coefficients and the reduced Diameter-to-Radius-of Gyration-ratio value for URDMO, it has been determined that the dynamic stability margins are satisfactory and that the degree of C.G. offset required to achieve the required L/D is not changed appreciably from that used on Viking.

ENTRY VEHICLE GEOMETRY AND MASS PROPERTIES COMPARED TO VIKING



Mass Properties

	URDMO	Viking
M (kg)	1.075×10^3 *	9.22×10^2
Ix(gm-M²)	2.187×10^6 *	7.335×10^5
Iy(gm-M²)	5.898×10^5 *	4.054×10^5
D (M)	3.505	3.505
S (M²)	9.65	9.65
D/σ	4.733	5.288

*Values in Lander Properties Table Supersede These Values

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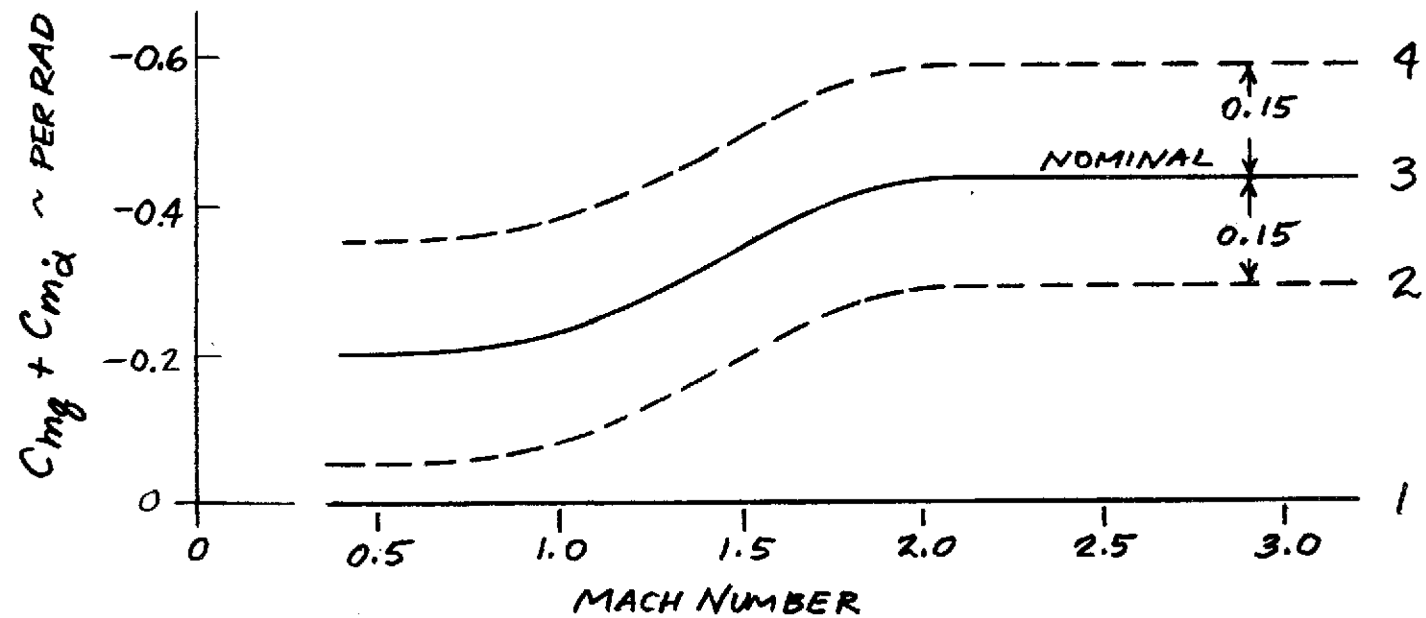
Pitch Damping Coefficient Functions



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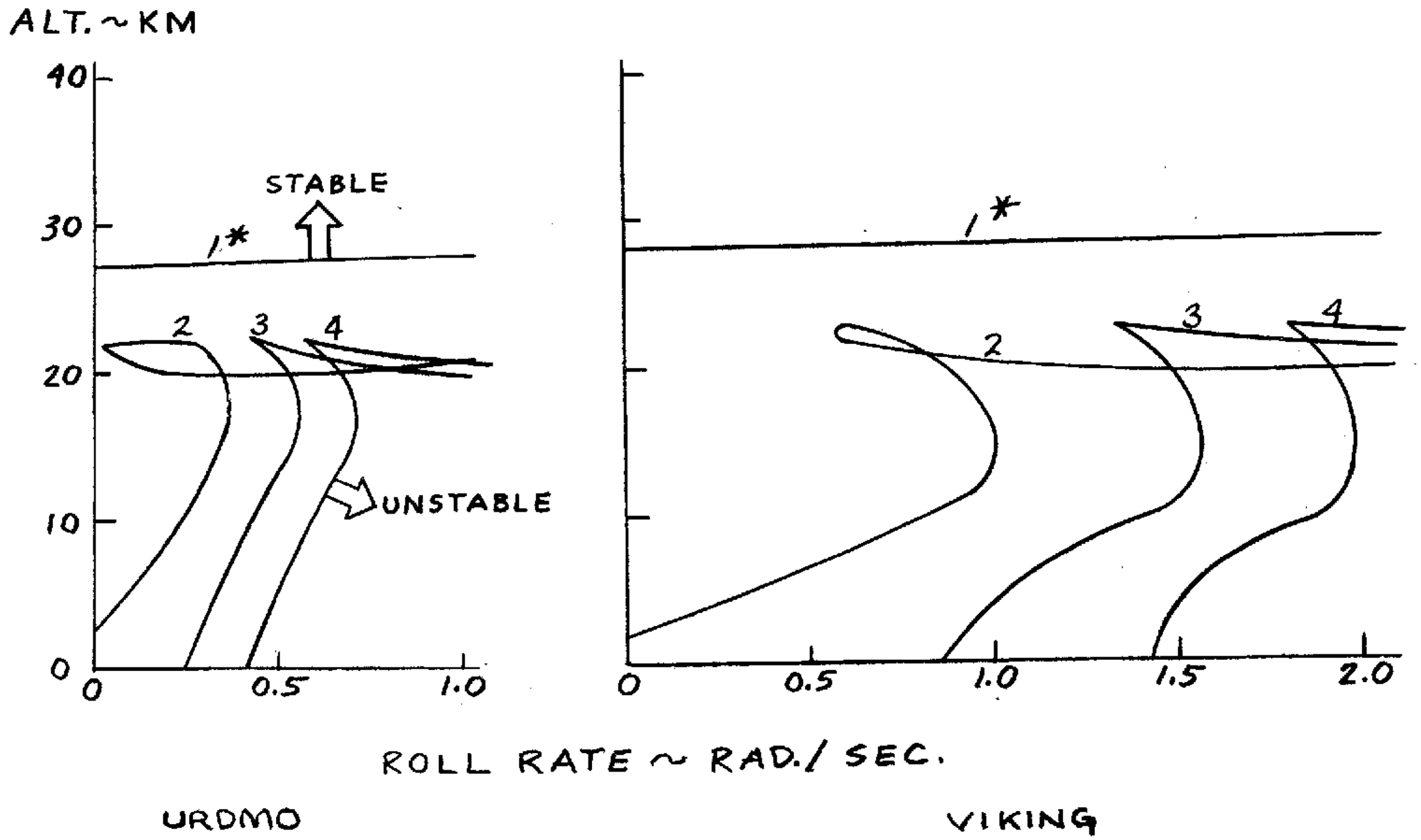
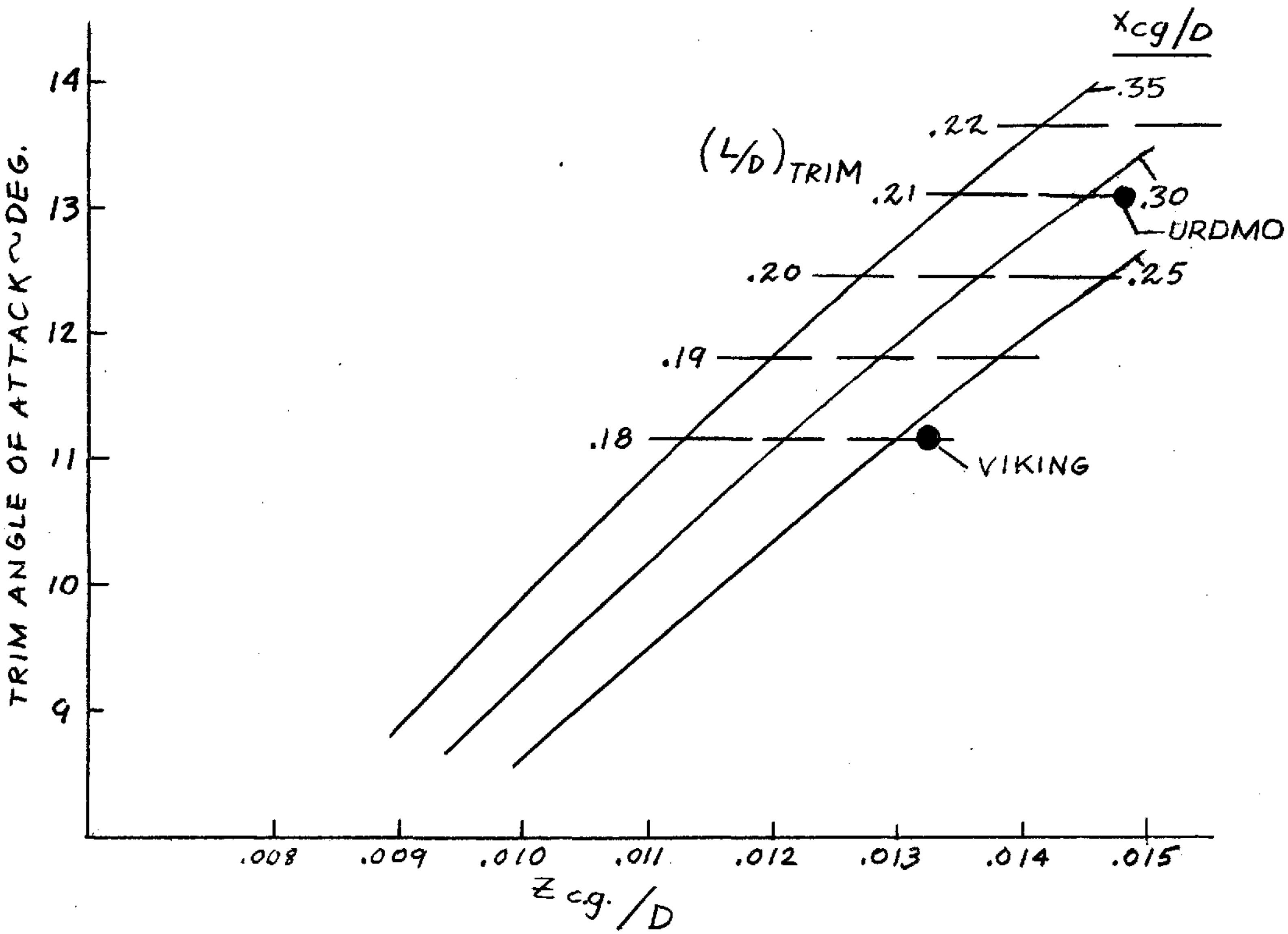


*NUMBER REFERS TO $C_{m_q} + C_{m_{\dot{\alpha}}}$ FUNCTION IN PREVIOUS FIGURE

Trim Angle of Attack for Various Center of Gravity Locations



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URDMO PROPULSION REQUIREMENT

The Viking orbiter and lander and earth return vehicle propulsion system concepts were only studied in sufficient depth to verify feasibility and establish weight. These systems did not require detailed analysis because they were either a slight modification of existing systems or were conceived from existing propulsion hardware. In either case, the function of usage of these propulsion systems was similar to that of the design basis.

The Mars Ascent Vehicle propulsion systems were analyzed in greater depth because of their functional usage and environment being different from that of existing propulsion systems. The energy requirement of a Mars ascent requires an efficient propulsion system with the design being further complicated by the requirement of sterilization. The selected concept of two solid motor stages and a third stage of monopropellant liquid propulsion, results from a combination of ascent trajectry and propulsion system studies. Recent development work has established that solid motors are sterilizable with a slight decrease in performance.

URDMO PROPULSION REQUIREMENTS

Propulsion System	System Requirement	Baseline Design
Viking Orbiter (Main	Trans-Mars Midcourse ΔV Mars Orbit Injection ΔV Mars Orbit Change MAV Rendezvous	Viking '75 Bipropellant Propulsion System with 14.5% Increase in Usable Propellant
Viking Orbiter (ACS)	Attitude Maintenance Attitude Change Assist in MAV Rendezvous	Hydrazine Monopropellant System Based on MJS
Viking Lander	Terminal Descent ΔV Roll Control	Viking '75 Monopropellant Propulsion with 31% Increase in Usable Propellant
Mars Ascent Vehicle	Mars Ascent Attitude Maintenance Rendezvous Assist	Two Stages of Solid Propellant and One Stage of Hydrazine Monopropellant
Earth Return Vehicle	Trans-Earth Injection Midcourse ΔV Attitude Stabilization and Change	Bipropellant Main Propulsion of VO'75 or Apollo Technology with Hydrazine or Cold Gas ACS

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ORBITER PROPULSION REQUIREMENTS

The primary requirements for the main orbiter propulsion system are like those of the VO 75 system. This results in the only change from Viking being a 14.5% increase in useable propellant. Previous studies have shown that it is feasible to increase the useable propellant by up to 60%. This increase will be achieved by adding a small additional barrel section to the propellant tanks.

The existing cold gas attitude control system does not meet the URDMO requirements in two areas. These are total impulse for the longer mission and control authority for the rendezvous with MAV. The higher thrust and total impulse requirements resulted in the definition of a hydrogen monopropellant system to perform the attitude control function. This system would operate in the blowdown mode with the engines being selected from the MAV, Viking, MJS or any of several existing spacecraft propulsion systems.

ORBITER PROPULSION REQUIREMENTS

Main Propulsion System

- Trans Mars Midcourse Corrections
- Mars Orbit Injection
- Mars Orbital Changes
- Mars Orbit Trims
- Rendezvous Closing Delta Velocity

Attitude Control Propulsion

- Separation Rate Reduction
- Limit Cycling
  - Trans Mars (299 days)
  - Orbital (440 days)
- Pointing Acquisitions
- Roll Searches
- Command Turns
- Main Engine Roll Control
- Terminal Rendezvous

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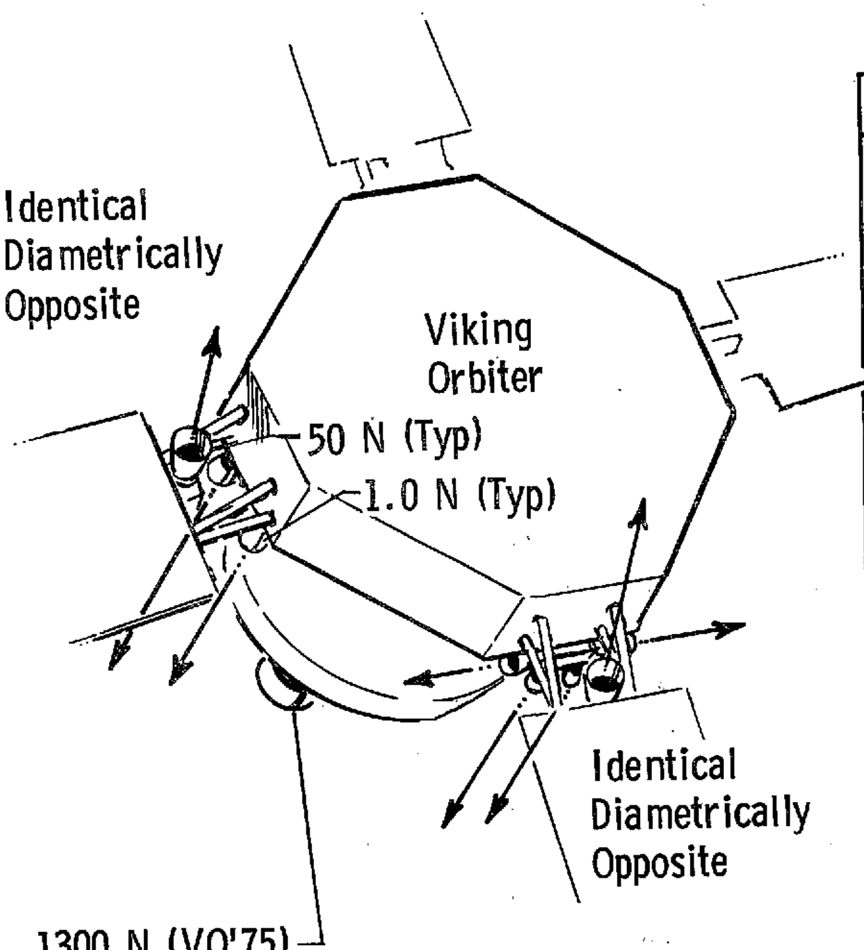
ORBITER PROPULSION SYSTEM

The facing page shows the modified Viking orbiter. The main purpose is to show the engine location that achieves the propulsion functions required of the orbiter. The primary attitude control function is supplied by the 12 - 0.5 to 1.0 N (0.1 to 0.2 lb) thrust engines arranged in the same manner as that planned for the MJS spacecraft and the Viking deorbit propulsion system. The forward facing 20 to 50 N (5 to 12 lb) thrust engine provides for extra thrust and high rate pitch/yaw maneuvers during docking with the MAV.

The 1300 N (300 lb) thrust engine (VO-75) provides for all other velocity change maneuvers, including the establishment of the closing velocity during initial and terminal rendezvous with the MAV.



Propulsion Characteristics

Main Propulsion
Bipropellant - N_2O_4/MMH 19% Stretched VO'75 Single-Gimballed 1300 N Thrust Engine Nitrogen Regulated Pressurization
Attitude Control Propulsion
Monopropellant - Hydrazine Derivative of MJS Propulsion Blowdown/GN_2 Pressurization 12 (0.5→1.0) N Thrust Engines 4 (20→50) N Thrust Engines

VIKING LANDER PROPULSION SYSTEM MODIFICATION

In order to accommodate the increased landed weight, the terminal descent and landing propulsion will require additional propellant and higher average thrust. Small increased propellant loads can be achieved by loading more propellant into the existing tankage and taking a higher blowdown ratio. The higher blowdown ratio results in a lower average thrust. This may be acceptable at the cost of reduced propulsion efficiency and would be an optimum solution for small landed weight increases.

Landed weight increases in the order of 100 kg will require both increased average thrust and propellant load. This can be achieved by one of the indicated methods. Additional study is required to determine the optimum solution. However, these data give a range of weight penalty associated with the increase in landed weight.

VIKING LANDER PROPULSION SYSTEM MODIFICATIONS

	Viking '75	Mod A	Mod B	Mod C
Tank Pressurization Mode	Blowdown	Blowdown	Regulated	Regulated
Thrust Profile*, N (lbf)	2670 1780 (600 400)	2670 1780 (600 400)	2670 (600)	2670 (600)
Usable Propellant, kg (lbm)	66.2 (146)	87.0 (192)	87.0 (192)	87.0 (192)
Modifications		Add Ullage Bottles & GN_2	Add Bottle, PCU & GN_2	Add Bottle, PCU & GN_2 Replace Prop. Tanks with Smaller Ones
Inert Weights				
Total GN_2, kg (lbm)	6.4 (14.0)	9.3 (20.5)	10.7 (23.5)	6.4 (14.0)
GN_2 Bottle(s), kg (lbm)		3.6 (8.0)	8.6 (19.0)	10.2 (22.5)
Press. Control, kg (lbm)			4.1 (9.0)	4.1 (9.0)
Prop. Tanks, kg (lbm)	14.7 (32.5)	14.7 (32.5)	14.7 (32.5)	8.2 (18.0)
Total, kg (lbm)	21.1 (46.5)	27.7 (61.0)	38.1 (84.0)	28.8 (63.5)
(Above V'75), kg (lbm)		6.6 (14.5)	17.0 (37.5)	7.7 (17.0)

*Available thrust (per thruster) from start to 90% propellant consumption (start of constant velocity descent).

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MAV PROPULSION SYSTEM CHARACTERISTICS

Propulsion requirements that evolve from the selected MAV mission profile consist of two large Delta V's (1654 and 2530 m/sec, respectively) to achieve a 100 x 2200 km orbit; smaller Delta V's (391 m/sec total) for orbit circularization, trim, and rendezvous; and attitude control and stabilization throughout the entire MAV mission. To satisfy these requirements, a 3-stage baseline propulsion system has been selected consisting of two solid propellant motors to provide the two large Delta V's, and a single monopropellant hydrazine system to provide the smaller Delta V's in addition to the attitude control functions during all phases of the MAV mission.

Solid motors were selected because of their superiority (high Isp and mass fraction) in the impulse range of interest to MAV. Their major limitations; inflexible configuration, lack of restart capability, high thrust-to-weight ratio, and non-sterilizability do not present problems for the MAV application, except for the latter one. Sterilizable solid propellants are not state-of-the-art, but are under development and should be available on a time scale compatible with the proposed Mars sample return mission.

Monopropellant hydrazine appears to be an ideal selection for the third stage propulsion system because of its comparative simplicity and high reliability, relatively high performance, and closely controllable impulse over an extremely wide range. It results in a relatively lightweight, compact installation.

MAV PROPULSION SYSTEM CHARACTERISTICS

Propulsion Requirements	Function	Baseline Selection
Delta V		
ΔV_1 1654 m/s	Ascent to 100 km	Solid Motors (2) High Isp, High λ Simple, Reliable
ΔV_2 2530 m/s	Transfer to 2200 km	
ΔV_3 341 m/s	Orbit Circularization	Monopropellant N_2H_4 Controllable Impulse Good Performance Simple, Reliable
ΔV_4 50 m/s	Trim/Rendezvous	
Attitude Control		
During Burns	Pitch Program Compensate Aero Moments Compensate Thrust Misalignment P, Y and R Stabilization	
During Coasts	Reorientation Maneuvers P, Y and R Stabilization	

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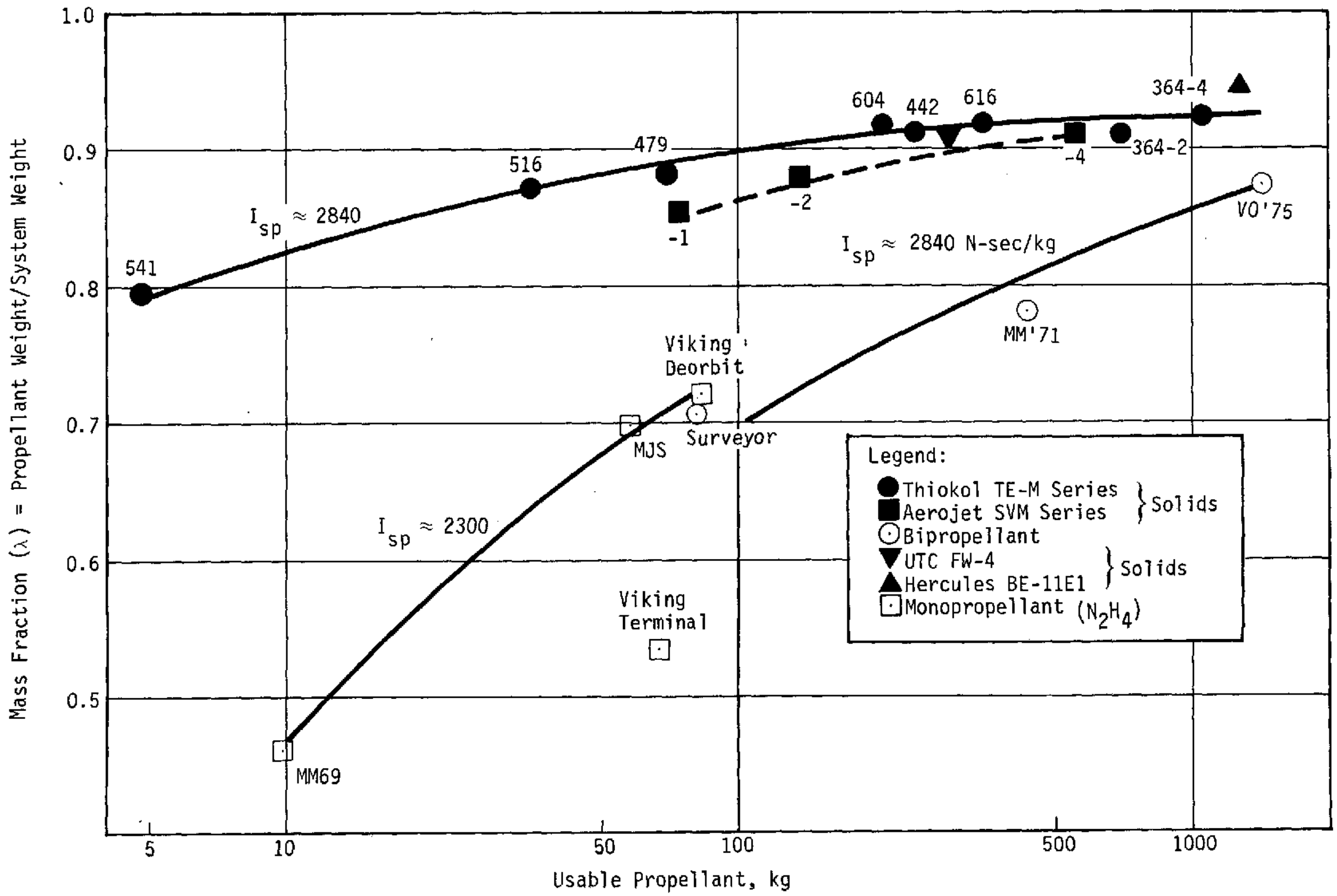
TYPICAL PROPULSION SYSTEM MASS FRACTIONS

The basis for the initial selection of the MAV baseline propulsion system is provided by the accompanying figure, a plot of system mass fraction versus propellant weight. For each of the two large Delta V burns associated with MAV ascent (requiring a propellant weight in the range of 100 Kg), it is evident that solid propellant motors are a logical choice if the principal consideration is performance (high Isp in combination with low weight). The solid motor provides Isp equivalent to that of earth storable liquid bipropellants, but is much lighter ($\lambda = 0.9$ versus 0.7 for bipropellants) in the impulse range of interest to MAV. Solid propellants do possess some limitations regarding flexibility of configuration and operation, but these are not detrimental in the MAV application. The bipropellant liquids become truly competitive only in sizes (total impulse) an order of magnitude larger than MAV.

It will also be noted that typical applications requiring multiple restarts and involving quantities of propellant less than about 100 Kg (220 lbm), utilize the monopropellant hydrazine system in preference to bipropellant systems because of its extreme simplicity and reliability. This provides a clue to the selection of monopropellant hydrazine for the MAV Stage III propulsion system.

TYPICAL PROPULSION SYSTEM MASS FRACTIONS



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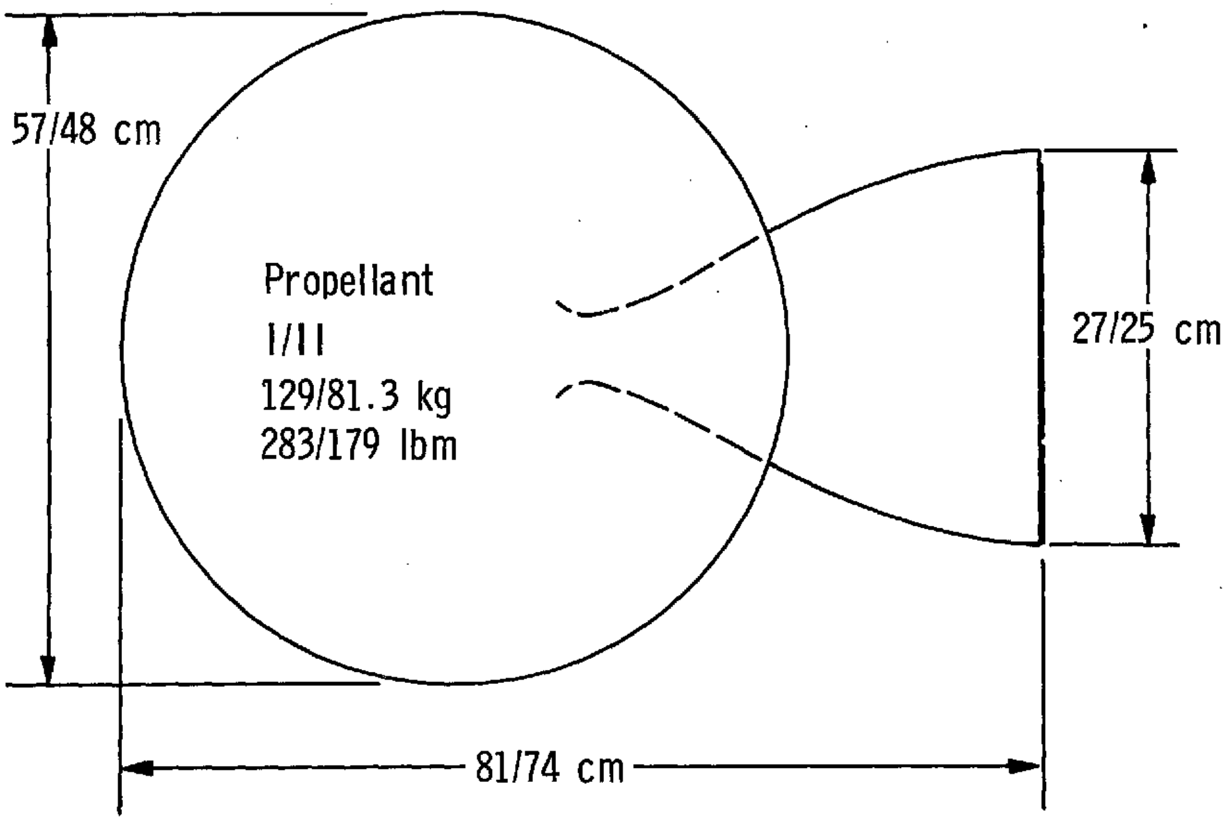
MAV SOLID MOTOR CHARACTERISTICS

Pertinent characteristics of the solid propellant motors selected for MAV Stages I and II are summarized in the accompanying figure. Both motors are of conventional spherical design and are fitted with submerged nozzles having an area ratio of 50 or greater. The Stage I motor is 57 cm (22 in.) in diameter and contains 129 kg (283 lbm) of propellant; Stage II is 48 cm (19 in.) in diameter and contains 81.3 kg (179 lbm) of propellant. The propellant formulation is not specified, but will be an aluminum - containing composite similar to the sterilizable formulations currently under development by JPL and AGC.

It is anticipated that both motors will operate at a chamber pressure of approximately 415 N/cm^2 (600 psia) and produce a thrust of 6675 N (1500 lbf). They will yield a specific impulse approximately two percent lower than current high performance solid propellants due to limitations imposed by the sterilization requirement, and will have slightly degraded mass fractions due to the requirement for a heavier liner to properly support the sterilizable propellant within the motor case.

MAV SOLID MOTOR CHARACTERISTICS



Isp	2973 N-s/kg 285 sec
λ	0.88
$\bar{F}$	6675 N 1500 lbf
t_b	55/35 sec
$\bar{Pc}$	414 N/cm^2 600 psia
A_e/A_t	50

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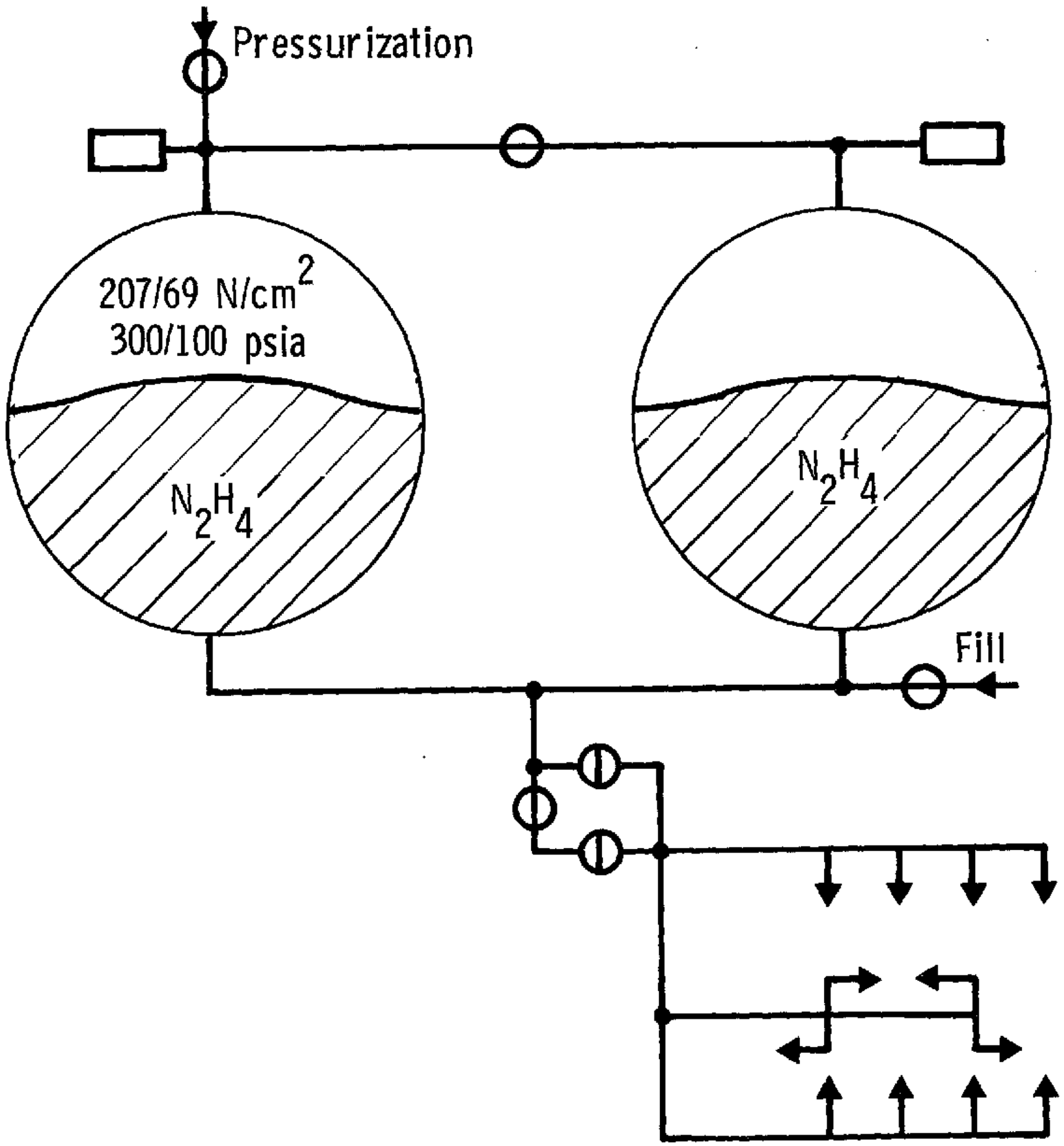
MAV STAGE III PROPULSION SYSTEM

Pertinent features of the proposed MAV Stage III propulsion system are summarized in the accompanying schematic. The system uses hydrazine propellant in the blowdown mode. Two propellant tanks approximately 23 cm ($9\frac{1}{4}$") in diameter are provided for packaging convenience. They contain bladders for effective propellant management in a zero-g environment, and are designed for sterilization following loading. Six pyro valves are used for propellant and pressurant loading and isolation functions.

It will be noted that a total of twelve thrusters are provided to perform all the required propulsion functions. Four aft firing thrusters (rated at 12 lbf each) will provide pitch and yaw control during Stage I and II operation, and will also provide the delta V requirement for orbit circularization, trim and rendezvous. Four tangential firing thrusters (rated at .4 lbf each) will provide roll control throughout the entire mission. Four forward firing thrusters (rated at .1 lbf each) will provide pitch and yaw stabilization during all Stage III coast phases.

The weight summary shows that the entire system is expected to weigh slightly less than 9 kg (20 lbm), a remarkably low value considering the many functions that the system is required to perform. This light weight is achieved partly through the elimination of redundant functions, except for arming (providing propellant to) the thrusters. Lack of redundancy is not a particularly desirable characteristic, but it is probably necessary because of the extreme weight limitations imposed on the MAV.

MAV STAGE III PROPULSION SYSTEM



Components	Weight kg (lbm)
Transducers (2)	0.2 (0.5)
Pyro Valves (2)	0.4 (0.9)
Plumbing	0.2 (0.5)
GN_2	0.1 (0.3)
Tanks (2)	1.9 (4.2)
Pyro Valves (4)	0.8 (1.8)
Plumbing	0.5 (1.0)
12 lbf Thrusters (4)	2.7 (6.0)
.4 lbf Thrusters (4)	1.1 (2.4)
.1 lbf Thrusters (4)	0.9 (2.0)
	8.8 (19.6)

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STAGE III THRUSTER SELECTION

The Stage III propulsion system has a multitude of functions to perform, but it has been determined that these can be satisfied by a total of only twelve thrusters. Four aft firing thrusters are required for pitch and yaw stabilization during Stage I and Stage II burns, and also to provide Stage III Delta V and pitch-yaw stabilization. Based on the maximum upsetting moments produced at Stage I burnout (maximum q), it is found that a thrust level of approximately 45N (10 lbf) is required. The Hamilton-Standard Model REA 22-4 thruster weighing about .7 Kg (1.5 lbm) is a logical candidate for this application.

Roll stabilization throughout the MAV mission is provided by four tangential firing thrusters. These must be large enough to provide adequate moments for roll stabilization during Stage I and II burns, but also must be capable of producing extremely small impulse bits so that propellant consumption during limit cycle operation is not excessive. A thrust level of approximately 2.2N (.5 lbf) is found to be acceptable, leading to the choice of the H-S Model REA 17-6 thruster as a logical candidate.

Pitch and yaw stabilization during Stage III coast periods is provided by four forward firing thrusters. These must be as small as possible to assure a low propellant consumption during limit cycle operation. A thrust level less than 0.5N (0.1 lbf) would be desirable, but catalytic hydrazine thrusters have not been developed in such small sizes. Therefore, the H-S REA 10-14 thruster rated at 1N (.2 lbf) is a tentative selection. It will provide acceptably low impulse bits.

It will be noted that total weight of the twelve thrusters is estimated to be 4.7 Kg (10.4 lbm).

STAGE III THRUSTER SELECTION

Configuration	4 Aft Firing	4 Tangential	4 Forward Firing
Function	P & Y Stabilization Stages I & II ΔV & PY Stability Stage III Burn	Roll Stabilization All 3 Stages	P & Y Stabilization Stage III
Sizing Criteria	Max q Stage I	Roll Moments Stages I & II Limit Cycle Stage III	Limit Cycle Stage III
Thrust Requirement	45(10) N (lbf)	2.2(0.5) N (lbf)	≤0.5 (0.1) N (lbf)
Candidates	REA 22-4 53 (12) N (lbf)	REA 17-6 1.8 (0.4) N (lbf)	REA 10-14 1 (0.2) N (lbf) (Throttled)
Weight (4)	2.7 (6.0) kg (lbm)	1.1 (2.4) kg (lbm)	0.9 (2.0) kg (lbm)

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MAV STAGE III PROPULSION SYSTEM DUTY CYCLE

The functioning of the Stage III propulsion system throughout the MAV mission is summarized in the accompanying table. The mission is considered to consist of seven major phases beginning with the Stage I burn, and ending with the orbit trim and rendezvous maneuver. Also indicated in the table are the approximate durations of each phase (ranging from 35 seconds to 400 hours), the Stage III thrusters that are operational during each phase, and the function that each thruster performs.

The final column of the table presents the estimates of the principal propellant usages during each phase. It will be noted that approximately 1.2 kg (3 lbm) of Stage III propellant may be consumed in providing stabilization during the Stage I and II burns, and .5 kg (1.1 lbm) is consumed to provide attitude control during Stage III coasts. The major usage is for the Stage III circularization burn which consumes 5.8 kg (12.6 lbm) of propellant. In addition .8 kg (1.7 lbm) is allocated for orbit trim and rendezvous maneuvers, and .8 kg (1.8 lbm) is allocated to cover propellant outage and contingencies. Total propellant required is approximately 9 kg (20 lbm). This value, combined with the Stage III propulsion inert weight of 9 kg (20 lbm) yields a Stage III mass fraction of .5, a relatively high value for such a small multi-purpose system.

MAV STAGE III PROPULSION SYSTEM DUTY CYCLE

Event	Time Interval	Stage III Thruster Operation lbf	Stage III Thruster Operation Function	Propellant kg (lbm)
1. Stage I Burn	55 sec	12.0	P-Y Control	0.9 (2.0)
		0.4	Roll Control	(0.1)
2. Coast	400 sec	12.0	Orient/Hold P-Y Attitude	
		0.4	Roll Stabilization	
3. Stage II Burn	35 sec	12.0	P-Y Control	0.3 (0.7)
		0.4	Roll Control	(0.1)
4. Coast, Elliptical	400 hrs	0.1	Earth Point/Hold, Reorient	0.2 (0.5)
		0.4	Roll Stabilization	0.1 (0.2)
5. Circularization Burn	100 sec	12.0*	ΔV (341 m/s)	5.8 (12.6)
			P-Y Control	
		0.4*	Roll Control	
6. Coast, Circular	70 hrs	0.1	Earth Point/Hold	**
		0.4	Roll Stabilization	
7. Rendezvous/Dock	3 hrs	0.1	Orbiter Point/Hold	0.8 (1.7)
		0.4	Roll Stabilization	
			Outage/Contingency	0.8 (1.8)
				8.9 (19.7)

* Thrust decays to 1/3 nominal value during burn.

** Included in 4. above.

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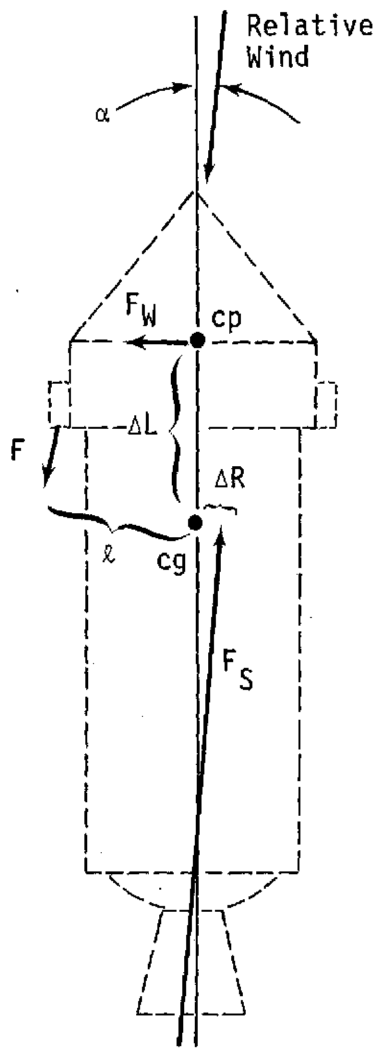
THRUSTER SIZING - AFT FIRING

The principal criteria for sizing the MAV baseline Stage III aft-firing thrusters are presented in the accompanying figure. The maximum moment that must be provided is associated with Stage I burnout when maximum q occurs. The combined aerodynamic and thrust misalignment moment is found to be approximately 1966 N-cm (174 lbf in.), necessitating a corrective thrust level of 48N (10.9 lbf). By comparison, the maximum pitch-over moment required during MAV ascent is negligible.

THRUSTER SIZING - AFT FIRING

Objective: Provide Adequate Moments for P-Y Control During Stage I & II Solid Motor Operation



Moments - Max q

Aero $M_W = C_{N\alpha}\, qS\, \Delta L$

$= 87.4\ lb_f\ in.$

Misalign $M_S = F_S\, \Delta R$

$= 150\ lb_f\ in.$

Resultant $M_{PY} = (M_W^2 + M_S^2)^{1/2}$

$= 174\ lb_f\ in.$

Required $F = \frac{M_{PY}}{\ell}$

$= 10.9\ lb_f$

Moments - Pitch Over

Inertial $M_I = I_o\, \alpha_{max}$

$= .94\ lb_f\ ft.$

$= 11.3\ lb_f\ in.$

Misalign $M_S = 150\ lb_f\ in.$

Resultant $M_{PY} = 150.4\ lb_f\ in.$

Assumptions

$C_N = .045$ deg

$\alpha = 1/3$ deg

$S = 3.7\ ft^2$

$q = 105$ psf

$L = 15$ in.

$F_S = 1500\ lb_f$

$\Delta R = .1$ in.

$\ell = 16$ in.

$I_o = 47\ slug\ ft^2$

$\alpha_{max} = .02$ rad/sec

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THRUSTER SIZING - ROLL

Pertinent criteria for sizing of the MAV baseline Stage III roll thrusters are presented in the accompanying figure. The thrusters must be large enough to compensate for aerodynamic roll moments and solid propellant exhaust torques, and yet be capable of providing very small impulse bits that consume a negligible amount of propellant during limit cycle operation. The maximum roll torque provided is based on Surveyor and Burner II experience, i.e., a torque of ~.025 Ncm (.01 lbf in.) is provided per Newton (lbf) of solid motor thrust. For MAV, this requirement evolves to a roll thrust level of ~2.2N (.5 lbf). Propellant consumption in the limit cycle mode is found to be only .1 Kg (.23 lbm), based on the assumption of a minimum impulse bit of .009 N-s (.002 lbf sec.).

THRUSTER SIZING - ROLL

Objectives: Provide Adequate Moment for Roll Control During Stage I & II Solid Motor Operation
Assure Propellant Consumption Not Excessive in Limit Cycle Mode (Stage III)

Solid Motor Burns

Surveyor Capability: $\frac{90 \text{ in. lbf Torque}}{9200 \text{ lbf Thrust}} \approx .01 \frac{\text{in. lbf}}{\text{lbf}}$

Usage: 25 in. lbf max.

Burner II Capability: $\frac{2(2.2)(26)}{9700} \approx .012 \frac{\text{in. lbf}}{\text{lbf}}$

3σ Duty Cycle: $\frac{1.55\ (65)}{2(2.2)(52)} = .44$

MAV Capability: $\frac{2(.4)(14)}{1500} \approx .0075 \frac{\text{in. lbf}}{\text{lbf}}$

Limit Cycle (Stage III)

$$\dot{w} = \frac{r\ (I_t)^2}{4\ \theta\ I_o\ Isp}$$

$$= 12.6\ (10)^{-8} \text{ lbm/sec}$$

W_p = .23 lbm (2 thrusters, 500 hrs)

Assumptions

r = 1.167 ft

θ = 10^o = .174 rad.

I_o = 1.758 slug ft^2

Isp = 120 sec.

I_t = .004 lbf sec (min) (2 thrusters)

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THRUSTER SIZING - FORWARD FIRING

Sizing of the MAV baseline Stage III forward firing attitude control thrusters is described in the accompanying figure. Consumption in the limit cycle mode is found to be .39 Kg (.86 lbm) based on a minimum impulse bit of .0022 N-s (.0005 lbf sec.) to be achieved by throttling the propellant flow with a Viscojet or similar device.

THRUSTER SIZING - FORWARD FIRING

Objective: Provide Acceptable Moments (Propellant Consumption) in Limit Cycle Mode

$$\dot{w} = \frac{r(I_t)^2}{4\theta I_o \; Isp}$$

$= .12\ (10)^{-6}$ lbm/sec (each axis)

$W_p = .22$ lbm (500 hrs, per axis)

$= .44$ lbm total

Assumptions

$r = 1.167$ ft

$I_t = .5\ (10)^{-3}$ lbf sec (throttled)

$\theta = 1/4^{\circ} = .00436$ rad.

$I_o = 1.16$ slug ft^2

$Isp = 120$ sec

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SOLID PROPELLANT STERILIZATION

The current generation of solid propellant motors will not satisfy the Viking sterilization requirements summarized in the accompanying table. The long soak periods at high temperature tend to produce excessive propellant decomposition with attendant formation of voids and/or cracks.

SOLID PROPELLANT STERILIZATION

Requirements: Component Qualification (Viking)

Two 54-hour Cycles at $125 \pm 2^{0}C$

Four 40-hour Cycles at $125 \pm 2^{0}C$

Flight Acceptance (Viking)

One 54-hour Cycle at $112 \pm 2^{0}C$

Propellant Development Contract

Six 53-hour Cycles at $135 \pm 2^{0}C$

Problem: Formation of Voids and/or Cracks Due to Decomposition and/or Differential Thermal Expansion at High Sterilization Temperatures.

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SOLID PROPELLANT STERILIZATION TECHNOLOGY

The state-of-the-art of solid propellant sterilization is summarized in the accompanying table. It will be noted that development programs are currently in progress at both JPL and AGC, with the firing of a full-scale sterilized motor being scheduled for February 1974. Reliable sterilizable motors producing a specific impulse of 2795 N-s/Kg (285 sec) and having a mass fraction of 0.88 are predicted for the late 1970's.

SOLID PROPELLANT STERILIZATION TECHNOLOGY

Early Investigations (JPL, NASA LRC, Thiokol, UTC and AGC):

Oxidizer Properties; Binders (Fuels); Subscale Firings;
Stress-free Support Concepts.

Current Investigations

JPL: ATS .71 m (28") dia. Apogee Motor, 364 kg (800 lbm) Saturethane Propellant
[81% Solids, 2695 N-s/kg (275 sec) Isp]

Stress-free Support (Silicone Fluid)

Loaded 12/73; Start Sterilization 2/74

AGC: Two .46 m (18") dia. SVM-3 Spherical Motors, 60 kg (133 lbm)
[ANB-3438, 84% Solids, 2795 N-s/kg (285 sec) Isp]

Stress Relieved Motor Concept
First Motor Developed One Small Crack (6 cycles at 135^{o}C)
Second Motor Successfully Completed Sterilization (8 cycles at 125^{o}C)
Firing Scheduled 2/21/74

Prediction for Late 1970s

Sterilizable Motor: Isp = 2795 N-s/kg (285 sec), λ = .88

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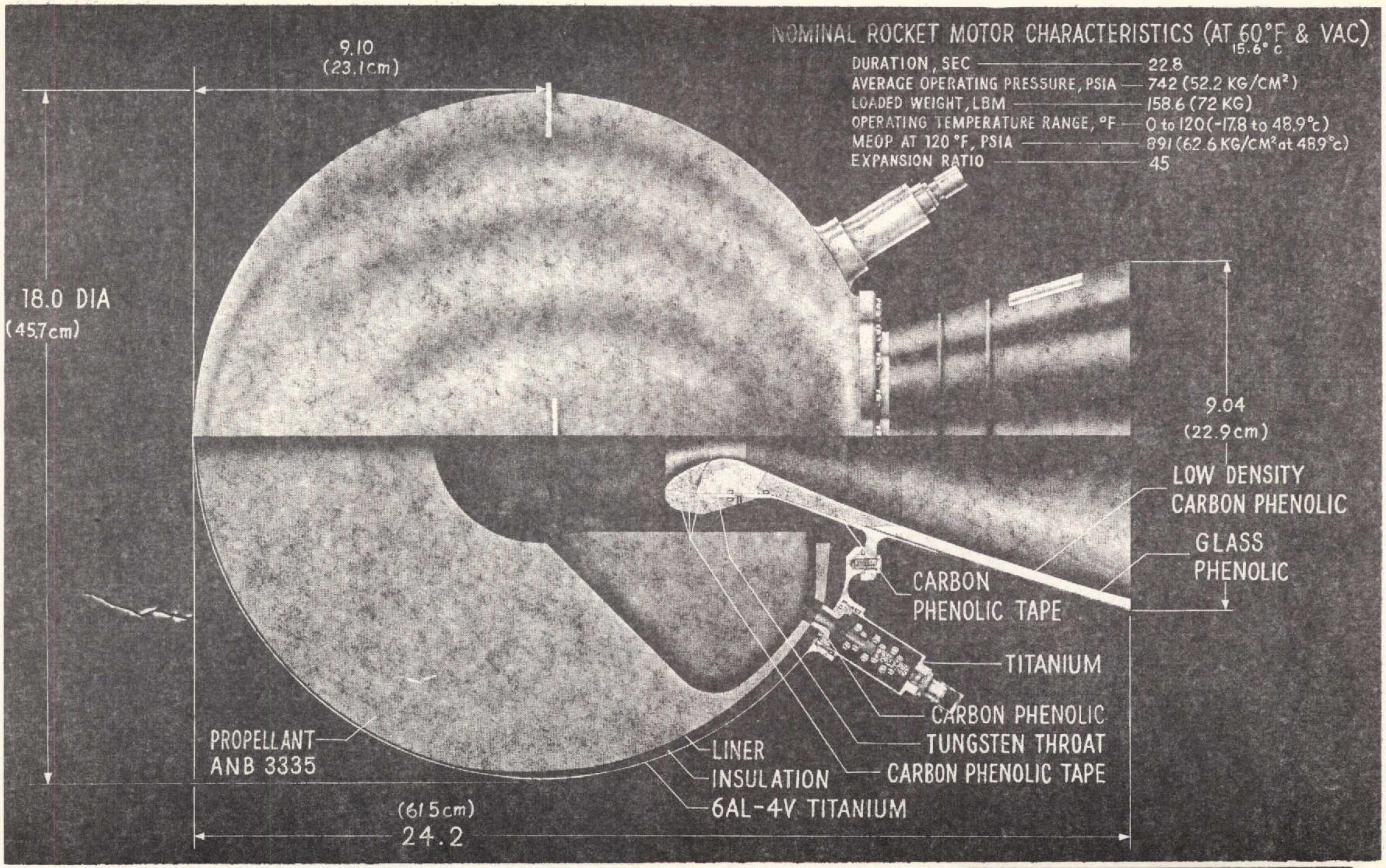
STERILIZABLE PROPELLANT DEVELOPMENT MOTOR

The first full-scale solid propellant motor to be fired following sterilization is the Aerojet SVM-3 motor shown in the accompanying figure. It is 46 cm (18 in.) in diameter and is loaded with 60 Kg (133 lbm) propellant of a special formulation. It has been subjected to eight sterilization cycles at 125°C, and is scheduled for firing in February 1974.

STERILIZABLE PROPELLANT DEVELOPMENT MOTOR



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BERYLLIUM - CONTAINING SOLID PROPELLANTS FOR MAV

One possibility for increasing the performance capability of the MAV solid propellant motors is to use Beryllium as a metal additive in the propellant instead of Aluminum. A specific impulse gain of 150 N-s/Kg (15 sec.) is attainable without experiencing any degradation of propellant physical properties. Because of the toxic nature of Beryllium, however, this approach does not appear to be very attractive. Qualification of such a motor would be extremely expensive, and the launch pad safety problems exceedingly difficult to resolve.

BERYLLIUM - CONTAINING SOLID PROPELLANTS FOR MAV

History:
- Loading/Firing Demonstrated by Solid Motor Companies
- Ecological/Toxicological Studies Conducted by -
  - RPL (Atmospheric Contamination)
  - AMRL (Effects of Animal Exposure)

Advantages:
- Isp Increase ~ 150 Nsec/kg (15 sec) vs Al
- Ballistic/Physical Properties Satisfactory

Limitations:
- Manufacturing Processes Very Costly
- Testing Costs Excessive
- Ecological Problems Almost Insurmountable

Conclusions:
- Technically Feasible, but --
- Ecologically Infeasible

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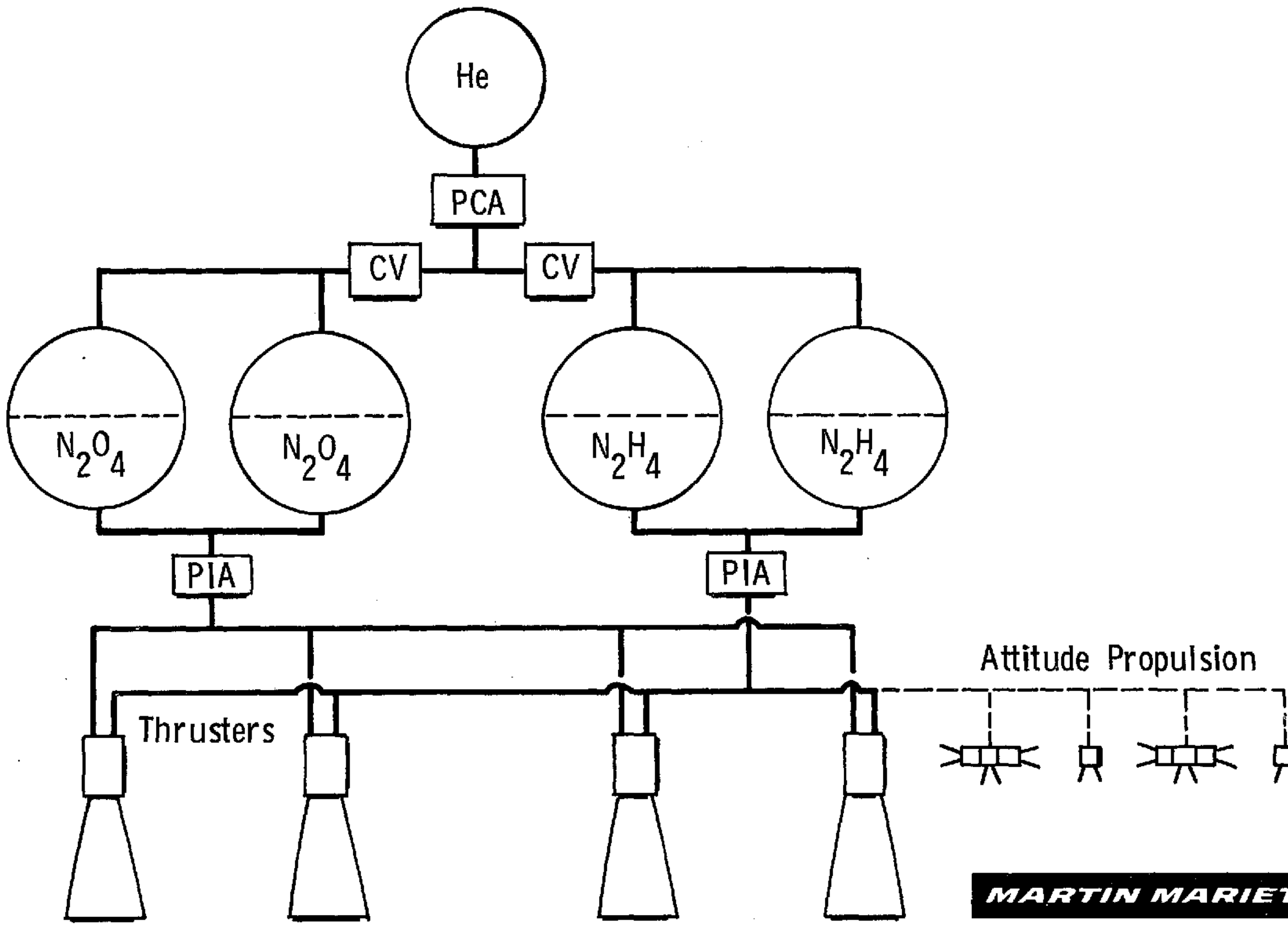
MAV BIPROPELLANT SYSTEM SCHEMATIC

One propulsion concept selected for comparison with the MAV baseline propulsion system is a combined Stage II and Stage III system using earth-storable bipropellants. This system, shown schematically in the accompanying figure, utilizes regulated helium for propellant tank pressurization, and includes four primary thrusters of approximately 445N (100 lbf) thrust each. The attitude control thrusters are small conventional monopropellant hydrazine thrusters.

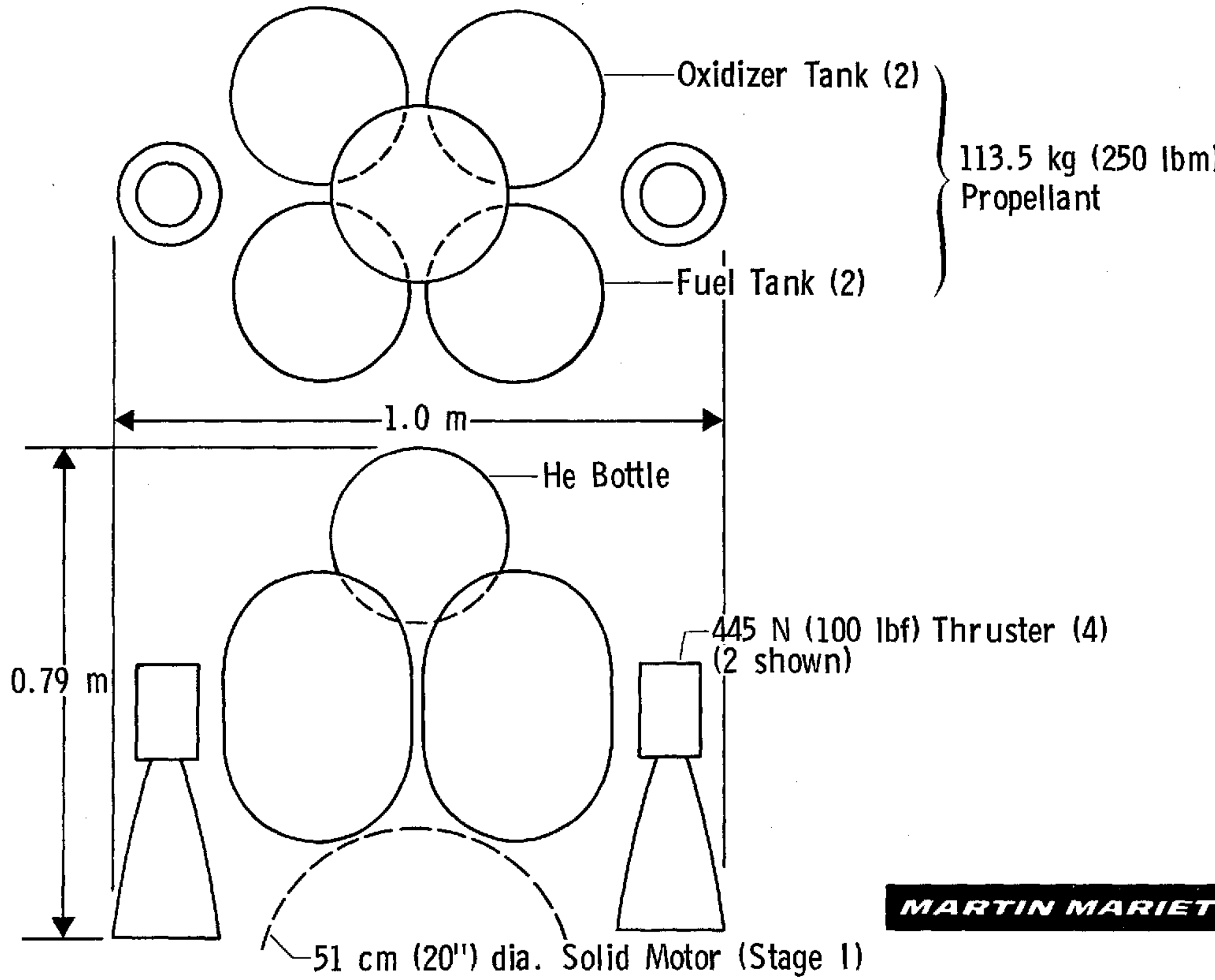
MAV BIPROPELLANT SYSTEM SCHEMATIC



MAV BIPROPELLANT STAGE

A possible configuration for the major components of the Stage II/III bi-propellant propulsion system is shown in the attached figure. This configuration provides a MAV that is somewhat shorter, but also wider, than the baseline configuration.

MAV BIPROPELLANT STAGE



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MAV BIPROPELLANT STAGE WEIGHTS

Weight estimates for the Stage II/III bipropellant system are presented in the accompanying table. Component weights are based principally on existing hardware associated with Apollo, Mariner '71 and the Viking Orbiter, and therefore are believed to be realistic. It will be noted that the propulsion inert weight totals to 48.3 Kg (105.9 lbm); residuals total 3.2 Kg (7.0 lbm). These, combined with the usable propellant weight of 113.5 Kg (250 lbm), result in a propulsion system mass fraction of 0.69.

MAV BIPROPELLANT STAGE WEIGHTS

Component	Capacity	Basis for Selection	Weight kg (lbm)
Propellant Tank (4)	207 N/cm^2; 28,000 cm^3 (ea) 300 psia; 1710 in^3	Apollo RCS, with Bladder	12.7 (28.0)
Pressurant Tank (1)	2620 N/cm^2; 9190 cm^3 (ea) 3800 psia; 560 in^3	VO'75 [9.2(10^6)N/cm^2 cm^3/kg tank] (370,000 psi in^3/lbm tank)	2.7 (6.0)
Thruster, Primary (4)	445 N (ea) 100 lbf	Apollo RCS (SM/LM)	9.1 (20.0)
Thruster, ACS (8)	2.2 N (ea) 0.5 lbf	H-S REA 17.6	2.2 (4.8)
PCA (1)		MM'71 (minus 2 pyros)	4.8 (10.5)
Check Assembly (2)		MM'71	2.2 (4.6)
PIA (2)		MM'71 (minus 4 pyros)	8.2 (18.0)
Tubing/Fittings		MM'71 + VO'75	6.4 (14.0)
			48.3 (105.9)
Propellant (N_2O_4/N_2H_4)	113.5 kg (usable) 250 lbm		
Propellant Residuals		MM'71 (2.5%)	2.8 (6.2)
Helium		VO'75 (He = 13% Press. Tank)	0.4 (0.8)
			3.2 (7.0)

$$\lambda = \frac{113.5}{113.5 + 48.3 + 3.2} = 0.69$$

MAV PERFORMANCE COMPARISON

A performance comparison of the baseline MAV and the alternate Stage II/III bipropellant version is presented in the accompanying table. Both versions are based on the same initial MAV weight of 250 Kg (550 lbm), and the same Delta V requirements. Both versions use the same Stage I solid propellant motor to provide the initial 1350 m/sec Delta V; the bipropellant Stage II/III then provides the remaining 3356 m/sec which, in the baseline version, is provided by two separate stages.

The resulting payloads (Stage III weight exclusive of propulsion inerts) are 20 Kg (44 lbm) and 2.7 Kg (6.0 lbm) for the baseline and combined Stage II/III, respectively, clearly showing the superiority of the former.

MAV PERFORMANCE COMPARISON

	Baseline MAV			Biprop. Stages II/III	
	Stage I	Stage II	Stage III	Stage I	Stage II/III
Propellant	Solid	Solid	N_2H_4	Solid	N_2O_4/N_2H_4
Isp. Nsec/kg (sec)	2795 (285)	2795 (285)	2300 (235)	2795 (285)	2892 (295)
Mass Fraction (λ)	0.88	0.88	0.50	0.88	0.70
ΔV, m/sec	1350	2865	491	1350	3356
Weight, kg (lbm)	113.0 (249)	104.5 (230)	32.2 (71.0)	113.0 (249)	137.0 (301.0)
Propellant	96.0 (211)	87.7 (193)	6.1 (13.5)	96.0 (211)	94.0 (206.5)
Prop. Inerts	13.2 (29)	12.2 (27)	6.1 (13.5)	13.2 (29)	40.3 (88.5)
Payload			20.0 (44.0)		2.7 (6.0)

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EARTH RETURN VEHICLE PROPULSION CONCEPT

The ERV propulsion systems have to provide the Trans-Earth velocity, midcourse corrections and attitude control. The Trans-Earth velocity will require a bipropellant propulsion system. The engine could either be the 1300 N (300 lb) thrust VO-75 engine or the 400 N (100 lb) thrust RCS engine from the Apollo Command Module and LEM. A separate cold gas system will be required for pointing and spin control. However, if either of the above engines could be qualified to use hydrazine as the fuel, these functions could use the monopropellant at a weight savings of approximately 10 kg (20 lbs).

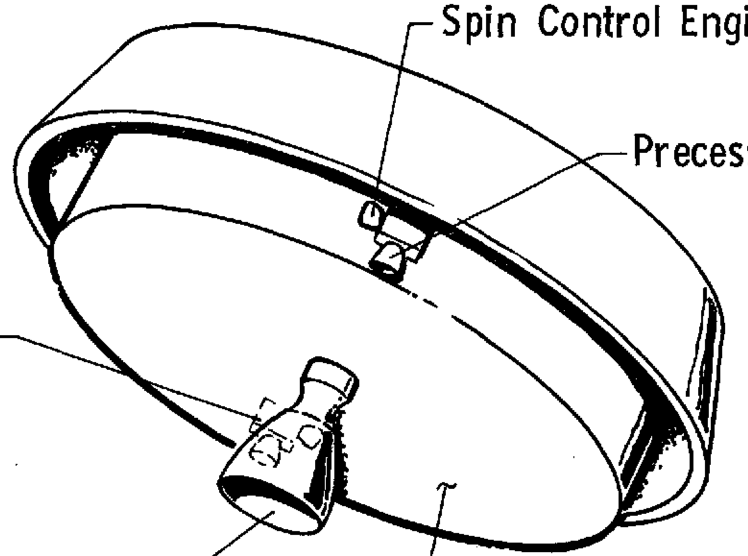
EARTH RETURN VEHICLE PROPULSION CONCEPT

Main Propulsion:

N_2O_4/MMH Bipropellant

Pressure Fed @ 100   150 N/cm^2

Thrust = 445 Newtons



Auxiliary Propulsion:

Cold Gas or Hydrazine

Thrust ≅ 20 Newtons

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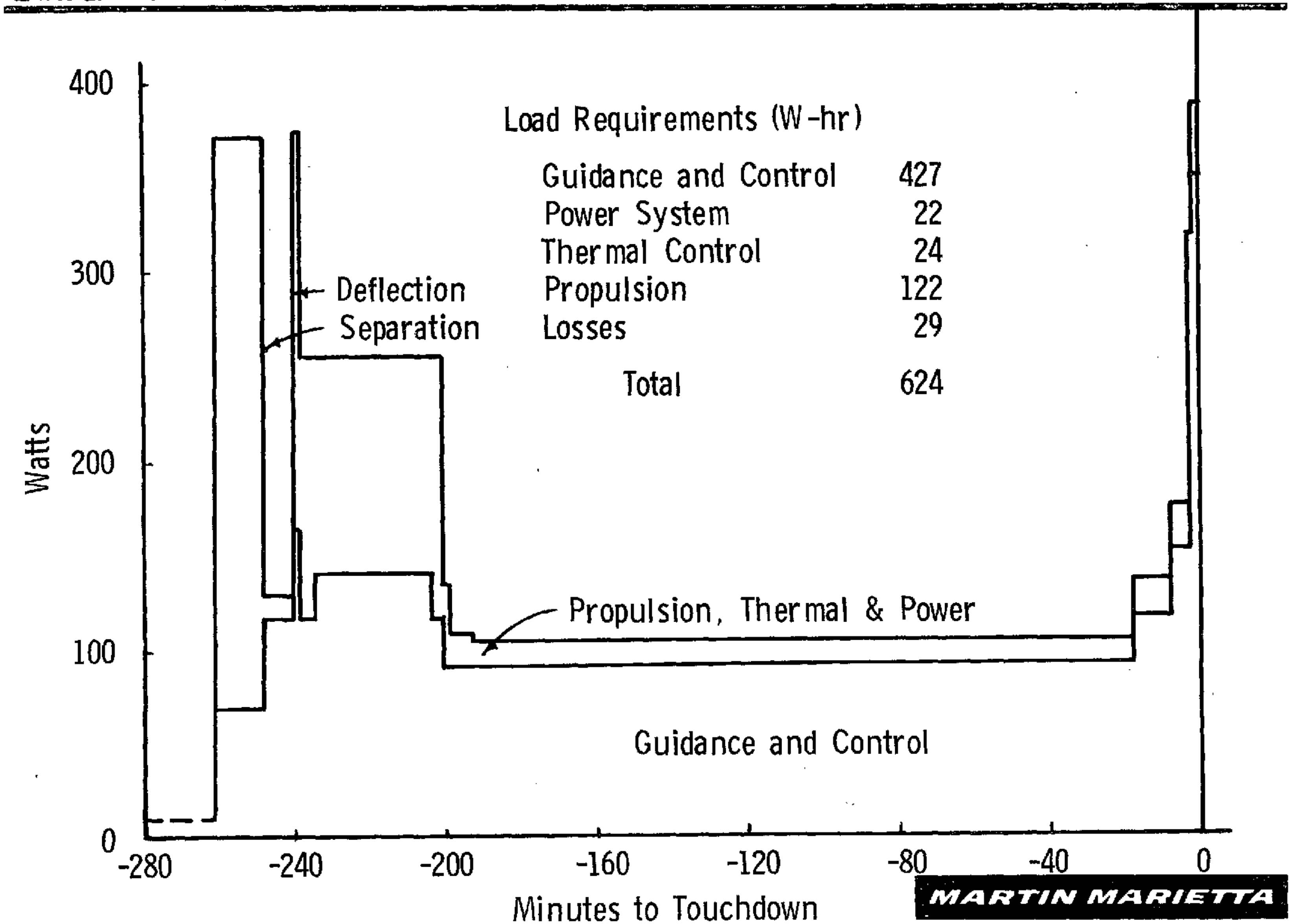
LANDER DEORBIT COAST ENERGY ALLOCATION

The deorbit coast energy allocation shown is based upon needs of the lander after transfer from the orbiter power system to the lander RTG/battery system. It is assumed that no communication, nor science equipment is in operation on the lander during the deorbit period. Tabulated are the watt-hours requirements based upon separation 250 minutes prior to touchdown. The operation schedule for guidance and control equipment together with that for propulsion engines is based upon Viking '75 timelines and power needs.

Power is to be provided by the RTG/battery subsystem. Two new RTGs using a selenide thermoelectric converter provide 20 watts each. When used with an 85% efficient converter 150 watt-hours of energy would be available after power transfer. This would be supplemented with 530 watt-hours of energy available from three 8-Ah nickel cadmium batteries (based on 75% depth of discharge). This is a sterilizable design of the type used in the Viking '75 Lander. The total available from both sources is then 680 watt-hours leaving a margin of 56 watt-hours after the energy needs for this phase of the mission are supplied.

LANDER DEORBIT COAST ENERGY ALLOCATION



LANDER POST-LANDED POWER ALLOCATION

Shown on this chart are tabulations of power and energy requirements for the Lander from touch-down to MAV liftoff. It is based upon the power profile shown on the next page.

The time on the surface is 263.75 hours. For the energy shown, this amounts to an average of 10.4 watts. In addition, provision must be made for thermal control of the Lander and the MAV together with a power margin for contingencies. In the case of Viking 75, this is 5 watts. All of these needs will be provided by the two 20-watt RTGs.

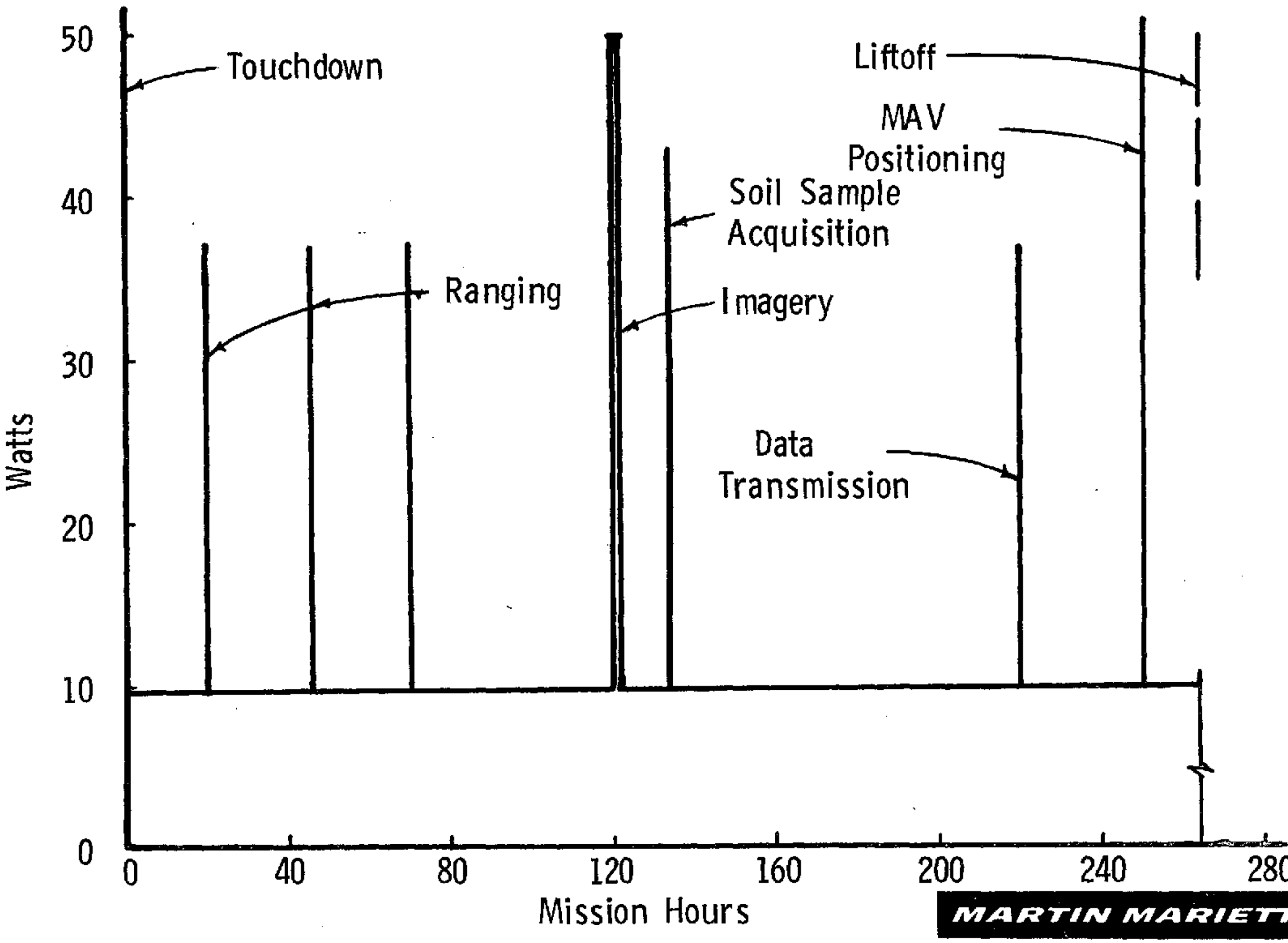
LANDER POST-LANDED POWER ALLOCATION

	Time (hrs)	Watts	W-Hrs
Continuous:	263.8		
S-Band Receiver (Pri.)	263.8	3.5	923.1
Command Detector	263.8	1.0	263.8
Command Decoder	263.8	0.5	131.9
GCSC	263.8	4.6	1213.3
Two-Way Communications			
S-Band Power Ampl.	7.0	13.0	91.0
S-Band Mod./Exciter	7.0	2.3	16.1
S-Band Receiver	7.0	3.5	24.5
Antenna Controller	7.0	2.0	14.0
Antenna Drive	7.0	0.6	4.2
Power Pie-Regulator	7.0	4.0	28.0
Telemetry Data Handling	7.0	2.0	14.0
Science			
Imagery	1.0	13.0	13.0
Soil Acquisition	0.1	33.4	3.3
MAV Positioning			
MAV Controller	0.1	2.0	0.2
MAV Drive	0.1	40.0	4.0
			2744.4

LANDED OPERATIONS POWER PROFILE

The power profile shown reflects the operation of the Lander from touchdown until MAV liftoff. Ranging using the S-band equipment is carried out on three successive days. Following this, a picture is taken and transmitted real time to Earth in order to select the area from which the soil sample is to be taken. After the soil sample is acquired, data verifying its acquisition is telemetered by the S-band channel to Earth and final commands for MAV positioning are received.

LANDED OPERATIONS POWER PROFILE



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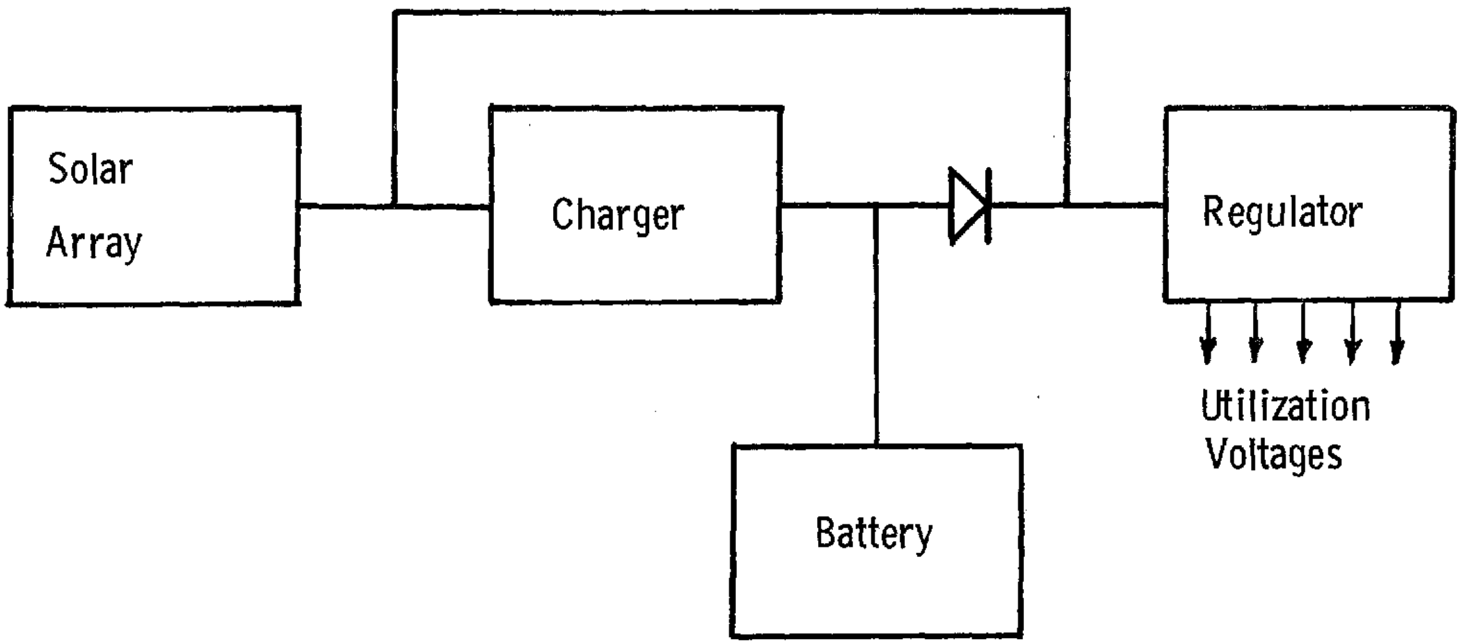
MAV POWER SUBSYSTEM BLOCK DIAGRAM

The energy shown in the previous tabulation is provided by a solar array battery system. Under the baseline condition, no sun occultation takes place during the MAV ascent. However, to accommodate other possibilities, an eclipse time equal to one-fourth of the orbit is used in sizing the solar array and battery. All of the utilization voltages including a-c power to drive the gyros is provided from the regulator block.

MAV POWER SUBSYSTEM BLOCK DIAGRAM

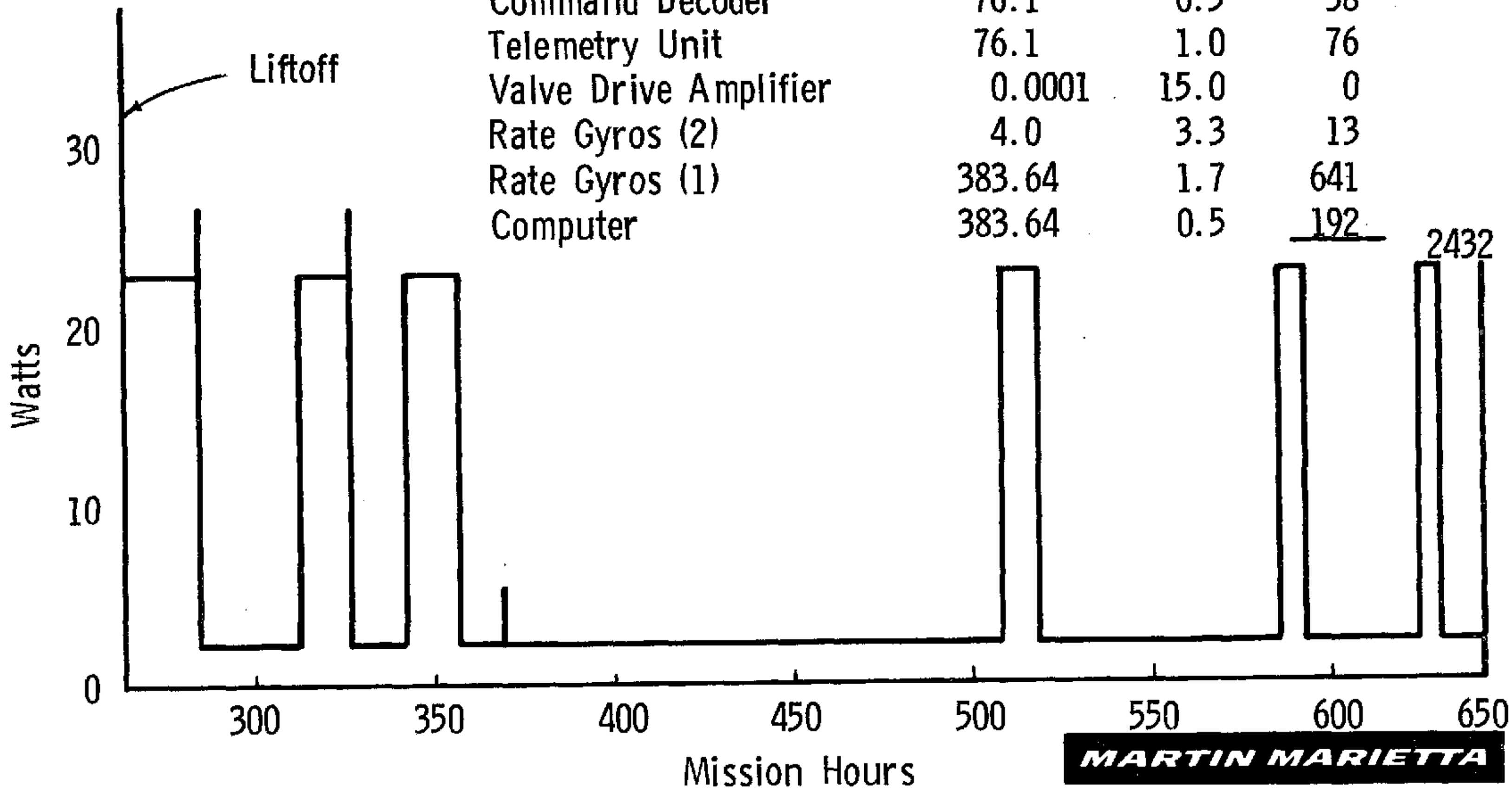


MAV POWER PROFILE

Depicted is the schedule of power demands resulting from the operation of equipment used in accomplishing MAV rendezvous with the Orbiter. The S-band transponder is used for doppler measurements that supply information for MAV trajectory correction. One gyro is in continuous operation to provide position information while the other two are operated only for the three midcourse corrections. The computer provides the sequencing from stored commands and updates received through the S-band receiver.

MAV POWER PROFILE

Item	Time (hrs)	Watts	W-Hrs
S-Band Power Amplifier	73.8	13.0	960
S-Band Modulator Exciter	73.8	2.3	170
S-Band Receiver	76.1	3.5	266
Command Detector	76.1	1.0	76
Command Decoder	76.1	0.5	38
Telemetry Unit	76.1	1.0	76
Valve Drive Amplifier	0.0001	15.0	0
Rate Gyros (2)	4.0	3.3	13
Rate Gyros (1)	383.64	1.7	641
Computer	383.64	0.5	192
			2432



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MAV POWER SUBSYSTEM EQUIPMENT LIST

Shown are ratings and masses of equipment items making up two power systems. One uses nickel-cadmium cells and the other nickel-hydrogen cells. Nickel-hydrogen cells are under development and promise to produce up to 100 Whr per kilogram. The regulator is sized to take care of the peak loads expected. The systems shown are of the lowest mass that may be expected and are based upon limiting the duration of peak load demands so as not to exceed battery capacity. This will require short periods of operation of S-band equipment, allowing time intervals for battery recharge. The average power used by the MAV is 6.4 watts. Battery charging and conversion losses will increase the power needed, requiring 10.5 watts to be supplied from the solar array for the minimum mass case. The necessary time line adjustment to be made is expected to allow the mass limits shown to be approached.

MAV POWER SUBSYSTEM EQUIPMENT LIST

Item	Rating	Wt (lb)	Mass (kg)
Nickel-Cadmium Cells	50.0 Wh	4.8	2.2
Regulator (Uncased)	42.0 W	2.2	1.0
Charger	11.0 W	1.2	0.5
Solar Array (without Substrate)	10.5 W	0.8	0.4
Harness and Connectors		1.3	0.6
Total		10.3	4.7
Option			
Nickel-Hydrogen Cell	50.0 Wh	1.5	0.7
Regulator (Uncased)	42.0 W	2.2	1.0
Charger	11.0 W	1.2	0.5
Solar Array (without Substrate)	10.5 W	0.8	0.4
Harness and Connectors		0.5	0.2
Total		6.2	2.8

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CANDIDATE EARTH RETURN VEHICLES

Several existing and new spacecraft candidates have been studied to determine their compatibility with the earth return mission phase. Dry spacecraft bus weight for each existing candidate was determined by removing excess capability, such as science and the associated power data handling, communications and power subsystems. Weight for these subsystems was replaced with weight of existing hardware that more nearly matched the ERV requirements. A bipropellant propulsion system was sized for the ERV velocity requirements. The resultant total spacecraft weight estimate is shown on the facing page.

None of the existing modified spacecraft can meet the present ERV weight allowance of 263 kg (578 lbs). One primary reason for the high weight of existing spacecraft versus a new ERV is the non optimum structural weight. This results in the ability to utilize existing components and technology to develop an ERV of spin or 3-axis attitude stabilization in the 200 to 250 kg (450 to 550 lb) weight class.

The round trip control module is also an attractive candidate from weight consideration. However, this candidate would require further study to assess the effects of cost, reliability and mission flexibility.

CANDIDATE EARTH RETURN VEHICLES

Candidate	Weight Relm, kg (lbs)	Attitude Stabilization	Comments
Mariner Venus/Mercury	500 (1100)	3-Axis	Existing-Out of Production
Pioneer Venus	350 (770)	Spun	Possibly Under Development
Pioneer 10/11	320 (700)	Spun	Existing-Out of Production
Mariner	600 (1300)	3-Axis	Existing-Out of Production
New 3-Axis (MAV Electronics)	225 (500)	3-Axis	Existing Technology
New Spun	225 (500)	Spun	Existing Technology
Round Trip Module	1300 (2900)	3-Axis	Possibly MJS Derivative

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VENUS PIONEER ERV CANDIDATE WEIGHT ESTIMATE

The facing page shows typical weight data for the Venus Pioneer modified for the ERV application. The basis for these weights was the multi-probe spacecraft. In addition to the weight modifications shown in the left column, the science package, attitude control system and propulsion system were removed from the spacecraft. Structural weight, directly associated with support of the science package, probes and propulsion, was also removed. The resultant spacecraft bus weight was 138 kg (306 lbs). The complete spacecraft would require the addition of propulsion and attitude control systems.

VENUS PIONEER ERV CANDIDATE WEIGHT ESTIMATE

	Weight, kg (lbs)		
Electrical Power	21.5	25.5	(56.2)
Added Solar Panels	+4.0		
Communications		13.2	(29.1)
Electrical Distribution	15.5	13.5	(30.0)
Less Removed Wiring	-2.0		
Data Handling		3.9	(8.6)
Thermal Control		15.5	(34.2)
Structure	75.4	61.7	(136.0)
Less Support for Science, Probes, etc.	-13.7		
Balance Weight Provision		5.4	(11.9)
Total Bus Dry Weight		138.4	(306.0)

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EARTH RETURN VEHICLE WEIGHT ESTIMATE

The 138 kg (305 lb) Venus Pioneer dry bus weight results in a total spacecraft weight of 388 kg (855 lb). The propulsion system was assumed to be bipropellant (N_2O_4/N_2H_4). The fuel (hydrazine) also supplied propellant for the spin and precession engines. A separate cold gas attitude control system would add approximately 10 kg (20 lbs) weight to the spacecraft total.

Starting with a total spacecraft weight allocation and subtracting propulsion system and payload weight results in a dry spacecraft bus weight of 88 kg (194 lb). This leaves a 50 kg disparity between Pioneer and the weight allocation.

Continuing studies will be required to determine the optimum ERV configuration that meets the weight allocation. The following candidates will be studied to determine the final proposed ERV design: 1) redesigned structure with Venus Pioneer Subsystems, 2) additional weight allocation or 3) new spacecraft utilizing existing technology to define the subsystems hardware.

EARTH RETURN VEHICLE WEIGHT ESTIMATE

Item	Pioneer Venus Candidate		Present Weight Allocation	
Spacecraft Bus Weight	138 kg	304 lb	88 kg	194 lb
Earth Entry Module	16	35	16	35
Soil Sample	1	2	1	1
Dry Weight Less Propulsion	155 kg	341 lb	105 kg	230 lb
Propellant (Usable)	189	417	128	282
Propellant Inerts	44	97	30	66
Total Spacecraft Weight	388 kg	855 lb	263 kg	578 lb

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THERMAL CONTROL PROBLEMS IN CURRENT STUDY

From sterilization through landing, thermal control is achieved by modified Viking '75 concepts, including the use of an RTG fluid loop during sterilization and prelaunch checkout, and passive thermal control from boost through landing on all equipment except propulsion. The modifications are required to accommodate the new RTGs which have different geometries and dissipate less heat at higher temperatures when compared to Viking '75 RTGs.

The most significant thermal problem during landed operations is maintaining the MAV propellant temperatures within the required limits. The MAV propulsion system is characterized by bulky geometry, no internal heat dissipation, narrow temperature limits, insignificant internal conductance, and relatively unprotected exposure to the Martian environments. The use of insulation and electrical heaters would be too heavy: the product of insulation weight and thermal watts required varies between 40 and 120 watt-power x kg insulation, from hot to cold extreme situations, respectively. A promising concept for the solution of both the heat source and heat distribution problems is depicted on the following vugraph. Thermal control of the MAV equipment compartment is achieved by the use of electrical heaters.

During Mars orbit, temperature control is achieved by passive means, including thermal coatings and insulation, as shown subsequently.

THERMAL CONTROL PROBLEMS IN CURRENT STUDY

Mission Phase	Problem Areas	Resolutions
Sterilization	RTG Heat Removal	Viking '75 Technology
Launch and Cruise	RTG Heat, "Initial Conditions" for Separation, Propulsion, Thermal Control	Modify Capsule Radiant Heat Distribution to Accommodate New RTGs, Use Viking '75 Technology
Separation Through Landing	Intense Internal and External Transients	Thermal Inertia - Viking '75 Technology
Landed Operations	Modified Lander Design; MAV Propellant Temp. Control = A Heat Distribution Problem	RTG Waste Heat Distributed Via Heat Pipes + Reflectors + Canopy; Electrical Heaters for MAV Compartment Temperature Control
Ascent/Docking/Orbit	MAV Equipment Comp. and Sample Canister Temperature Control	Control Achieved by Thermal Inertia and Passive Thermal Control

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MAV PROPELLANT TEMPERATURE CONDITIONING DURING LANDED OPERATIONS

The "canopy" concept uses RTG waste heat as a source for thermal control, supplied in the form of "line-sources" via heat pipes. The heat pipe temperatures will be between 170 and 250°C. Radiant heat from the heat pipes will be directed essentially upward by IR reflectors (polished aluminum) as shown, and the radiation will be re-reflected and distributed around the MAV propulsion system by the reflective finish on the interior surfaces of the canopy. The "gap" between the canopy and the MAV serves as an insulator with effective conductivity = conductivity of Martian atmosphere + convective effects. Data obtained during the investigation of convective coupling between the outer shell and the LN_2 shroud of a large thermal vacuum chamber (29 x 65 ft) when operated at Martian pressure levels indicate that the convective effects inside the canopy should be acceptable.

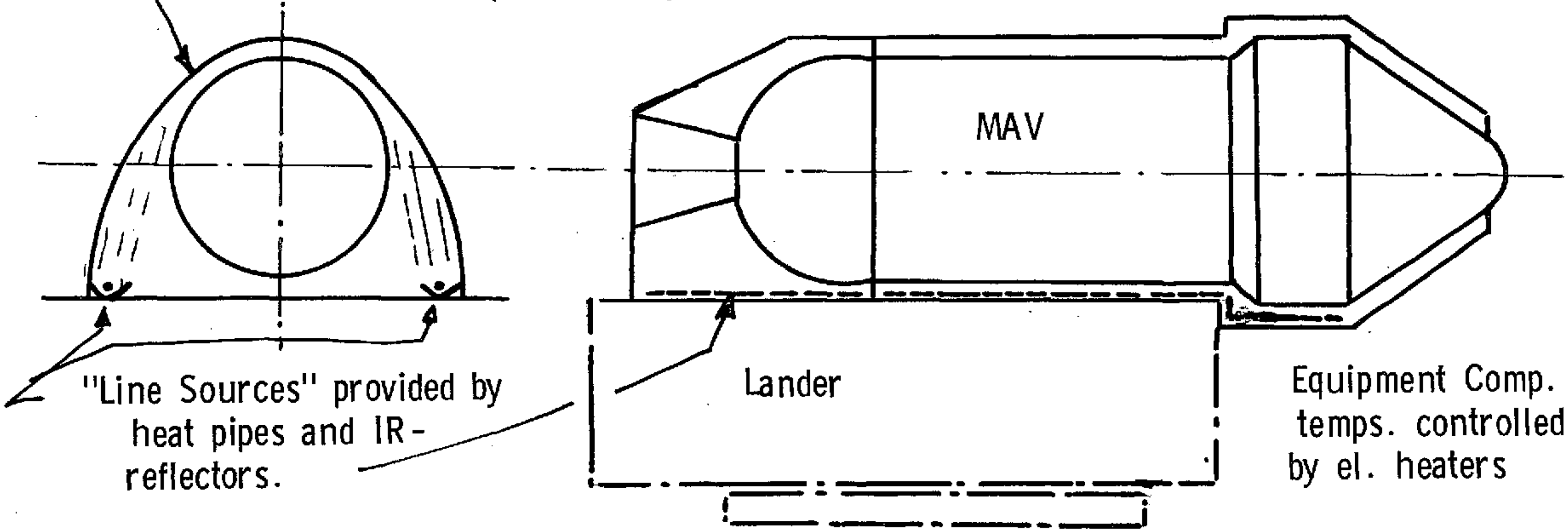
Control to accommodate hot and cold extremes is achieved in one or a combination of three possible ways: (1) rotation of the reflectors around the axes of the heat pipes via bimetallic actuators or equivalent; (2) size the system to survive the hot extreme, compensate for cold extremes by electrical heaters; (3) size the system for an appropriate nominal environment and qualify propulsion system for the hot and/or cold extremes.

The concept requires verification by test.

MAV PROPELLANT TEMPERATURE CONDITIONING

"Canopy" concept minimizes insulation/heater requirements :

* IR-reflective internal finish provides radiant heat distribution from heat pipes, minimizes heat loss to environment
* Gap between canopy and MAV provides insulation - with convective losses compensated by heat from heat pipes



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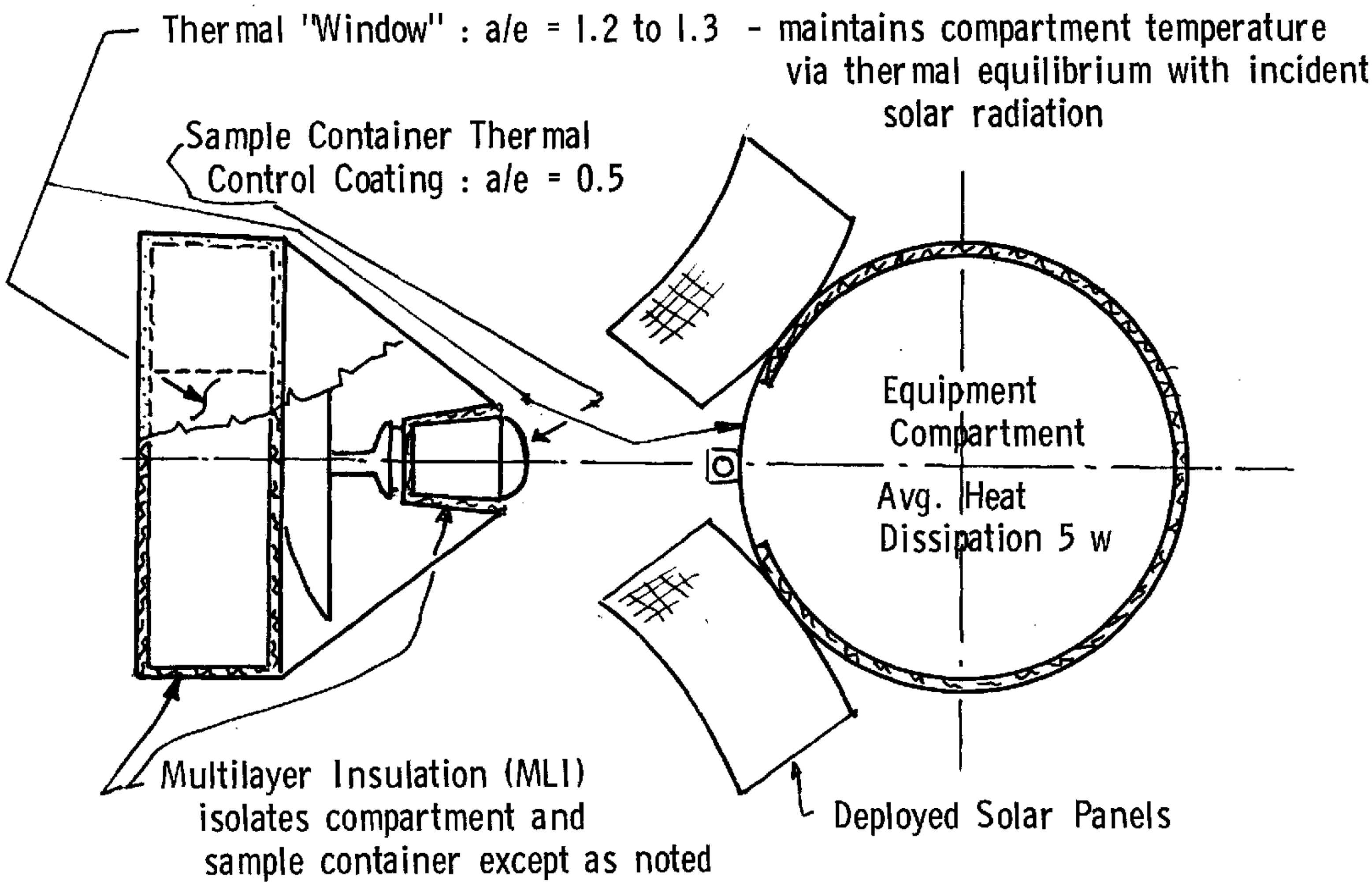
MAV THERMAL CONTROL DURING ORBITAL OPERATIONS

This concept takes advantage of the constant solar orientation of the MAV. Equipment compartment temperatures are maintained by passive thermal balance between the absorbed solar and emitted IR radiation through the "thermal window". The interior of the compartment is thermally coupled to the "window" by radiation, and it is thermally isolated from the rest of the spacecraft and from the space environment by multilayer insulation (except the window).

A similar concept is used to control the temperature of the sample container, with an absorptivity/emissivity ratio of $a/e = 0.5$, in order to maintain its temperature below $0^{\circ}C$. The solar angle was assumed constant at 35 degrees from the vehicle axis.

MAV THERMAL CONTROL DURING ORBITAL OPERATIONS



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REMAINING STUDY TASKS

W. T. Scofield

SUGGESTED SECOND HALF STUDY TASKS

The tasks shown here are planned for the next three months of the study effort. They can be modified or substituted for at the discretion of the JPL Technical Manager.

Additional work on the MAV Stage III subsystems will involve development of functional design requirements as well as possible improvements in the propulsion telecommunications, power, guidance and control and structural subsystems.

SUGGESTED SECOND HALF STUDY TASKS

Improved MAV Stage III Subsystems

Impact of Increased Sample Size

More Details on Orbiter Mods

Circular vs Eccentric Rendezvous Orbit

Spin Stable vs Three Axis Stable MAV

Additional Navigation Analysis

Additional Rendezvous and Docking Analysis

Landing Latitude Accessibility

Analysis of Backup and Redundant Mission Features

1981 vs 1983/84 Mission Requirements

Detailed Mission Profile (Task 3.5)

Technology and Programmatic Assessment (Task 3.6)

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POTENTIAL FOLLOW-ON TASKS

These tasks have been identified so far as being pertinent to the understanding of the MSSR mission but outside the scope of the current study.

POTENTIAL ADD-ON TASKS

Design of Earth Entry Capsule to Minimize Back Contamination

Conceptual Design of Earth Return Vehicle

Round-trip Control Module vs Orbiter + ERV

Improved Sampling Techniques

Additional Orbiter and Lander Science

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